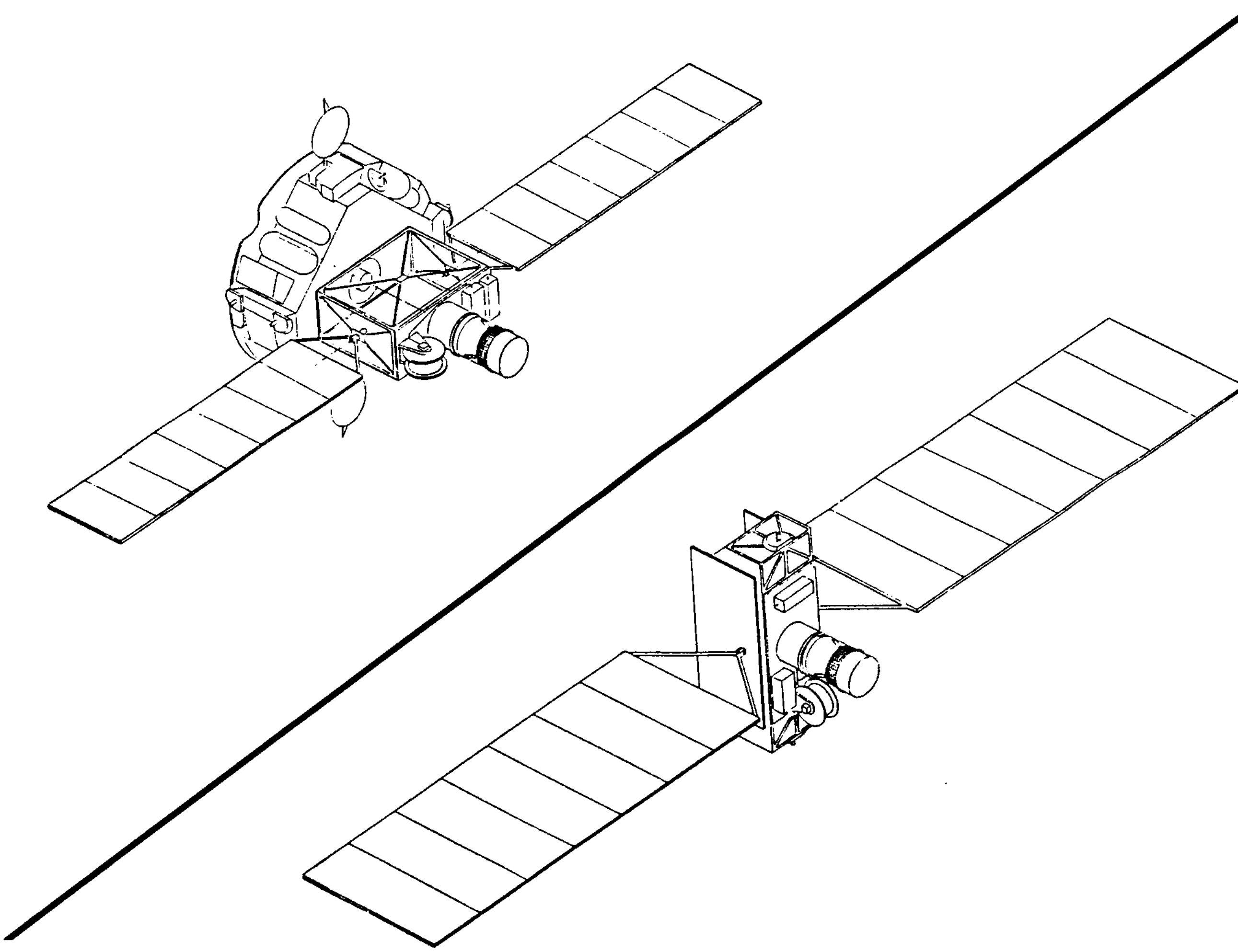
Study of Plasma Motor Generator (PMG) Tether System for Orbit Reboost

Final Report

September 1987



TRW

Study of Plasma Motor Generator (PMG) Tether System for Orbit Reboost

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September 1987

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FINAL REPORT PMG TETHER STUDY

1. INTRODUCTION AND STUDY PLAN

The emerging technology of electrodynamic tethers promises substantial benefits for America's space program. To date, however, the feasibility and advantages of integrating tethers with spacecraft remain largely in the realm of theory. NASA's study of Plasma Motor Generator (PMG) tether systems for orbit reshoot is an important step in establishing the validity of this applied technology.

This report describes the results of a study by TRW for NASA on a 2 kW PMG tether system to be used for orbit reboost. This study represented an opportunity to advance the state of the art of a new space technology. Successful demonstration should lead to the inclusion of such a system on future spacecraft as an integral part of the propulsion and electrical power systems.

The objectives of the study were to:

1. Develop a viable 2 kW PMG engineering design model
2. Select representative missions and spacecraft from current mission models for possible inclusion of a PMG tether system
3. Perform studies to assess the impact of integrating the PMG tether system with the candidate spacecraft
4. Study the operations of the integrated systems
5. Produce performance figures as a basis of selecting a single mission and spacecraft
6. Create an engineering design and development plan for the PMG tether system tailored to the selected mission.

Figure 1-1 is the study plan for these activities.

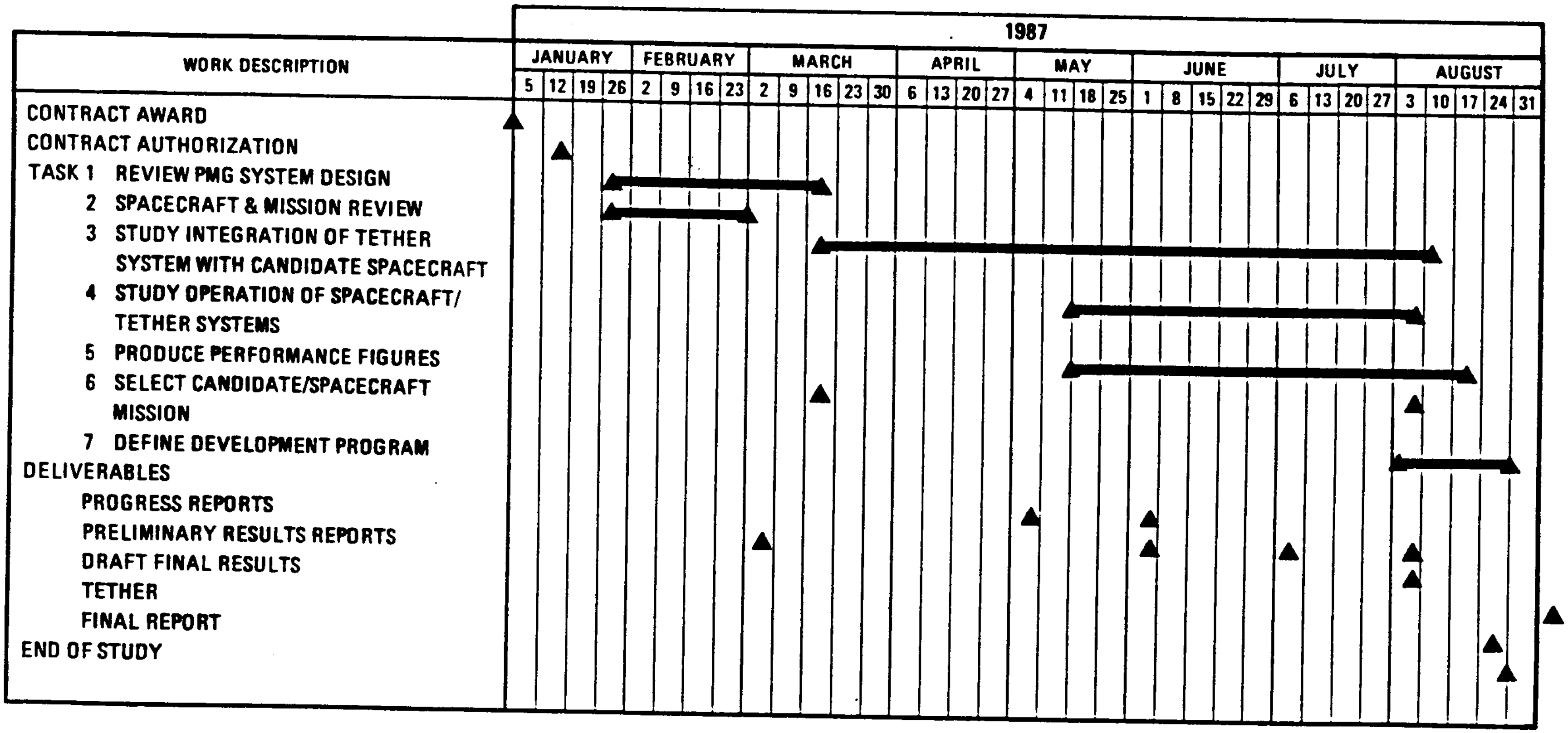

Figure 1-1. PMG Tether System Study Plan

2. ELECTRODYNAMIC TETHER THEORY AND TECHNOLOGY

In this section, simple explanations of tether theory and relevant technology are presented as the basis of understanding the PMG tether system.

Consider a conducting tether wire of length $\underline{L}$ cutting across the geomagnetic field $\underline{B}$ with orbital velocity $\underline{v}$. An open-circuit potential difference V is generated between the ends of the tether given by

$$V = (\underline{v} \times \underline{B}) \cdot \underline{L} .$$

This potential difference induces a current i to flow through the tether causing an electrodynamic force given by

$$\underline{F}_d = i\underline{L} \times \underline{B} .$$

This force is opposite to the velocity, producing power at the expense of orbital energy.

The tether can also be used for orbit reboost by reversing the current flow in the tether. If enough current is supplied to the tether by a spacecraft to overcome the induced current so that a net current flows in the reverse direction, an accelerating force will be produced causing an altitude increase at the expense of electrical power from the spacecraft.

For the tether to function as a power generator or an orbit reboost motor, a current loop must be completed through the ambient plasma of the ionosphere. The connection between the tether ends and the ionsphere is made via a plasma contactor. The baseline contactor design is a hollow cathode. It is preferred over the alternatives of passive collectors and electron guns because of weight, size and power advantages. The tether concept using a hollow cathode as the plasma contactor is referred to as the PMG tether system.

The hollow cathode works by flowing argon or xenon through the cathode tube. The external heater causes thermionic electron emission from the insert which in turn creates a plasma within the cathode tube. The anode draws electrons from the cathode creating a plasma ball outside the cathode tube. The plasma ball functions as the plasma contactor, supplying whatever charged species is required to complete the current loop.

Tether technology includes the tether material, storage and deployment mechanisms, tether control and stability, and dynamic interaction with the spacecraft. The recommendations for these mechanisms and findings on these issues constitute the major portion of this report.

3. SELECTION OF HOST SPACECRAFT

The search for suitable spacecraft to host the flight demonstration of the PMG tether system began with the mission models of NASA Johnson Space Center, the Space Assembly, Maintenance and Servicing Study (SAMSS) and the Orbiting Maneuvering Vehicle (OMV) program. Many missions were eliminated because the orbits did not cross geomagnetic field lines, for example those flying in polar orbits. Others were eliminated because the plasma and magnetic field are too weak, for example those in geosynchronous orbits. Experiments attached to the Space Station or to the Shuttle were discarded because of the lack of need for orbit reboost. The remaining missions were categorized into ten groups based on host spacecraft and function as required by the statement of work for more careful examination. Table 3-1 lists these candidates.

These candidate spacecraft and missions were reduced to four; OMV, ISF, Spartan and RPDP after additional information was gathered and more stringent criteria applied. These criteria included launch before 1998, the need for orbit reboost, the design of the host spacecraft being at a stage conducive to the integration of the tether system, ample power, and not having a stringent pointing accuracy requirement. Figure 3-1 summarizes the missions rejected and the reasons.

In addition to the four missions listed above, the European Retrievable Carrier (EURECA) was included in the final list because of its highly desirable characteristics. These are mission length, ample power, design as an all-purpose payload host, and early availability. For the purposes of this study, two of the five spacecraft, OMV and EURECA, were selected.

Table 3-1. Candidate List for Host of Flight Demonstration

Spacecraft/Function Category	Mission(s)
Space Station	Tethered Electrodynamic Power
Space Telescope	Large Deployable Reflector Planetary Multispectral Telescope
Earth Observation	Microwave Remote Sensing Geology Satellite
Upper Atmosphere Platform	Recoverable Plasma Diagnostic Package (RPDP) Space Station Coorbiting Platform Explorer Astronomical Platform Systems Technology
Power Supply	All
Industrial	Industrial Space Facility (ISF) Space Processing for Advanced Materials Commercial Space Processing Materials Processing Lab
Life Sciences	CELSS Biotechnology
Technology Demonstration	Orbiting Maneuvering Vehicle (OMV) Spartan

MISSION \ REASON FOR REJECTION	HIGH POINTING ACCURACY	VERY HIGH MASS	LOW POWER	NOT A FREE FLYER	COMPETING EXPERIMENTS	LATE LAUNCH OR AVAILABILITY	POSSIBLE INAPPROPRIATE ORBIT	INSUFFICIENT INFORMATION
LARGE DEPLOYABLE REFLECTOR	X	X						
EXPLORER PLATFORMS	X		X					
PLANETARY MULTISPECTRAL TELESCOPE	X							
MICROWAVE REMOTE SENSING	X		X					
CELSS				X				
BIOTECHNOLOGY				X				
SPACE PROCESSING OF ADVANCED MATERIALS				X				
COMMERCIAL SPACE PROCESSING TEST				X				
ASTRONOMICAL PLATFORM	X					X		
PLATFORM SYSTEMS TECHNOLOGY						X	X	
TETHER SYSTEM (ESA)					X	X		
MATERIALS PROCESSING LAB				X				
TETHERED ELECTRODYNAMIC POWER GENERATOR		X			X	X		
GEOLOGY SATELLITE								X
SPACE STATION CO-ORBITING PLATFORM						X		X

Figure 3-1. Candidate Missions and Spacecraft Rejected

4. PLASMA CONTACTOR EQUIPMENT

Early work on the concept of a PMG tether system is contained in References 1 and 2. This design served as a starting point for this study.

This section will discuss the plasma contactor requirements for the PMG Tether System. The topics will include the power and mass requirements, the plasma contactor configuration and the requirement for a contactor at the host end of the system. In Sections 5 and 6, the tether materials and design and electronics are discussed in detail.

4.1 PLASMA CONTACTOR DESIGN

The plasma contactor design is based on the hollow cathode design that has been used in plasma contactor research (References 3 and 4). The design is shown schematically in Figure 4-1. The device consists of a tantalum tube electron beam welded to a thoriated tungsten orifice plate. Located within the hollow cathode is a thin rolled tantalum foil insert which is electrically connected to the cathode tube. The inner surface of the insert is coated with a low work function material (barium and strontium carbonates). Wrapped around the cathode tube is heater wire. The heater is composed of a resistive wire filment insulated from the tantalum casing. A radiation shield of tantalum foil is wrapped around the heater wire. Several millimeters downstream of the cathode is a disc anode (called a keeper by those familiar with hollow cathode technology) used to initiate and sustain hollow cathode operation.

For a flight design the tantalum foil insert should be replaced by a hollow porous tungsten plug. Experience with foil inserts has shown that the foil deteriorates, becoming brittle and flaking after short periods. This substitution will result in a slight increase in the hollow cathode power requirement.

The expellant gas recommended is xenon. The use of xenon will increase both the lifetime and operating efficiency of the contactor over that of argon, an alternate gas considered as an expellant.

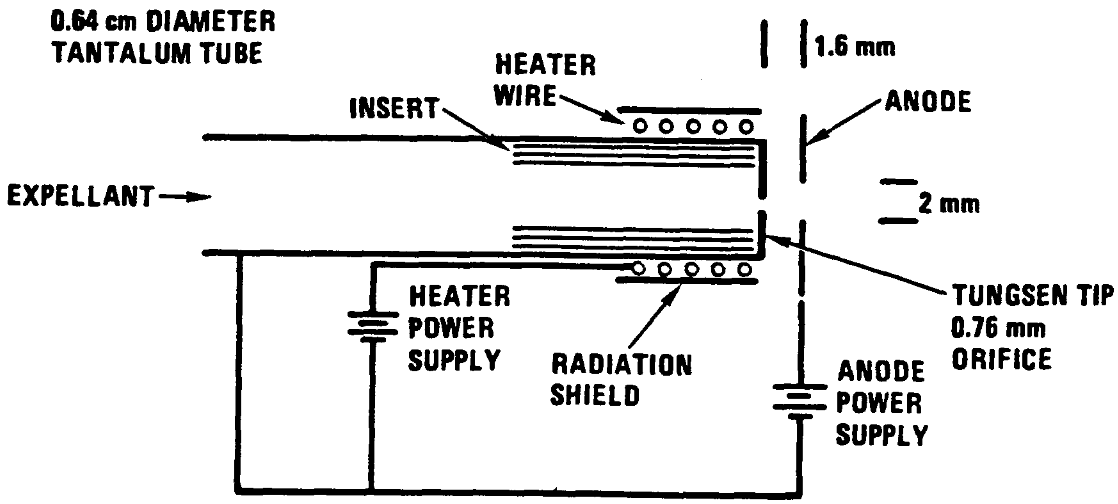

Figure 4-1. Schematic of Hollow Cathode

4.2 PLASMA CONTACTOR REQUIREMENTS

The plasma contactor equipment requirements include the hollow cathode power supplies as well as expellant storage and control hardware. The hollow cathode will require two power supplies, the heater and anode supplies, as shown in Figure 4-1. In addition, a high pressure tank (4200 psi) is required for storage and a redundant pressure regulated system is needed for expellant flow control. A heater will be required for the storage tank to maintain the expellant above its critical temperature 17°C for xenon). A redundant plasma contactor assembly is recommended for reliability.

The expellant supply system weight and mass estimates are given in Table 4-1. The mass of the hardware is based on space-qualified hardware (References 5 and 6). The expellant storage tanks are spherical with a diameter of 35 cm. The weight of the xenon is based on a flow rate of 5 sccm. The weight and mass estimates for the hollow cathode power supplies are given in the section on electronics. The sizing of the expellant

Table 4-1. Weight and Mass Estimates for Expellant Supply System (per contactor)

	Weight (lb)	Mass (kg)
Expellant tankage	6.0	3.0
Xenon supply (1 year)	31.0	14.0
Regulation system	8.5	3.9

tankage heater supply must await a thermal analysis. The mass of the hollow cathode itself is negligible. The hollow cathode is approximately 7 cm long.

4.3 PLASMA CONTACTOR POWER REQUIREMENTS

The contactor power requirements includes the heater power, the power to operate the hollow cathode discharge (anode power) and the power lost in the contactor sheath potential drop (supplied by the tether). The power requirements are based on the plasma contactor performance curve shown in Figure 4-2 (Reference 3). The total power requirement is 127 W. A breakdown of the power requirements is given in Table 4-2. Similar performance figures have been achieved with the slightly different cathode design from Johnson Space Center (JSC).

The power estimate assumed that a plasma contactor is required at both ends of the tether system. The validity of this assumption will be demonstrated later in this section.

4.4 CATHODE POISONING

Once a cathode has been operating, the rare-earth oxide coating on the inner surface of the insert is susceptible to contamination by oxygen and moisture. At operational altitudes, contamination should not be a problem due to the low concentration of oxygen and the expellant flow. The concern is over cathode poisoning occurring between the cathode ground tests and launch. Using a hollow cathode without the oxide coating would require considerably more heater power (Reference 7) and is therefore not

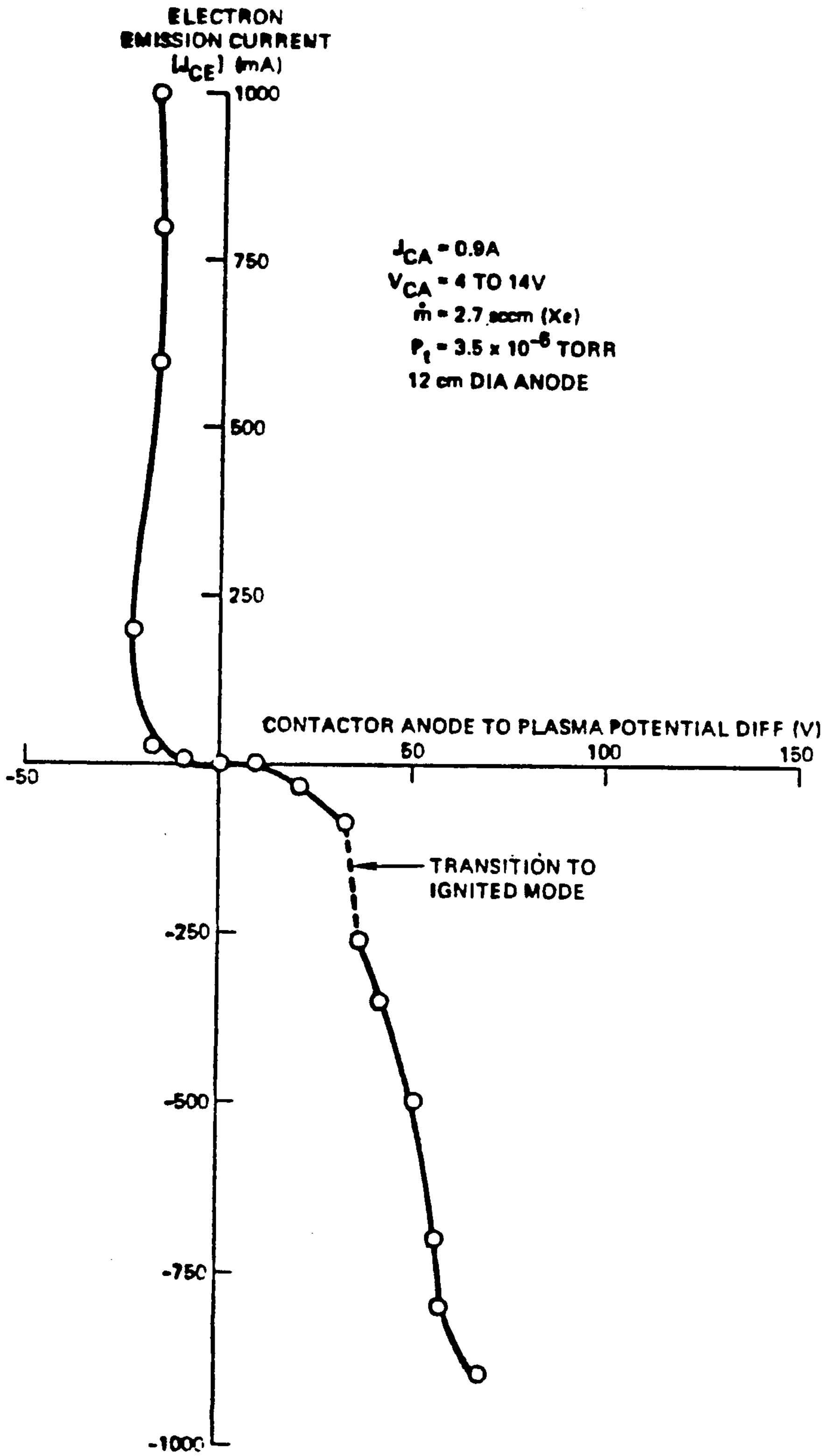

Figure 4-2. Conventional Contactor Performance

Table 4-2. Plasma Contactor Power Requirement

	Watts
1 A electron emission	
Heater power	15
Discharge power	13
Sheath power	20
1 A electron collection	
Heater power	15
Discharge power	4
Sheath power	60

suggested. A controlled enclosure preventing contamination of the oxide between the ground test and launch is recommended.

Another approach would be the use of an internal starter electrode. The starter electrode would initiate a discharge within the cathode tube (Reference 7). Once the discharge has been started, the starter electrode would be turned off. The electrode would allow easier start-up but would require a more complex contactor with an additional power supply. This approach has been suggested for other applications of hollow cathodes (Reference 8) and is presently employed in the plasma contactor research at JSC.

4.5 PLASMA CONTACTOR REQUIREMENT FOR HOST END OF TETHER SYSTEM

It was assumed when estimating the contactor power requirements that a contactor would be required on both ends of the tether system. This analysis validated that assumption. A plasma contactor on the host end is needed if the spacecraft structure cannot collect from the environment the required current of 1 A required by the tether system. The current collection is dependent on the plasma environment, spacecraft velocity and spacecraft structure potential, relative to the ionsphere.

The plasma environment is one of the determining parameters in calculating the current collection. Since the tether system will fly in low earth orbit (LEO), the analysis was confined to altitudes between 300 and 1000 km. The plasma parameters used were obtained from Reference 9. The plasma parameters (density, temperature, and ion species) vary with altitude as well as with time of day. The ion density not only varies with time of day but also with the dominant ion species. At night hydrogen is the major ion species over these altitudes, while during the day atomic oxygen is the major ion species.

As the tether system is reversible, the spacecraft end must be able to collect both electron and ion currents. The thermal current density measure is A/m^2 and for electrons is three orders of magnitude greater than ions. As such, the ion collection capability of the spacecraft structure is much lower, and is therefore the limiting collection process. Only the ion collection was calculated for this analysis.

The ion collection is dependent on the spacecraft structure potential. In the 2 kW system, a potential different of 2 kV will exist between the spacecraft host and the tether end mass. How the host potential will equilibrate with the plasma would require another analysis. For these calculations, the two extreme points, 0 and 2 kV, was considered.

At zero potential, the ion collection will be dominated by ram collection since the ram flux is 6 to 35 times greater than the thermal flux. The ram ion current collection is dependent on ion density and spacecraft velocity. The required surface area for one ampere of ram ion collection is shown in Figure 4-3(a).

At 2 kV structure potential, the ion collection will be predominately thermal ions and will be controlled by sheath effects. To date, an accurate model of ion collection at high voltages has not been developed. An empirical model of the ion collection by a plate based on experiments on large structures (Reference 10) has been used to calculate the ion collection. The predicted spacecraft structure area required to collect one ampere of thermal ion current is shown in Figure 4-3(b).

The calculations of the surface area of the host spacecraft structure required for the ion collection demonstrated that the required area is so large that it makes this collection scheme impractical.

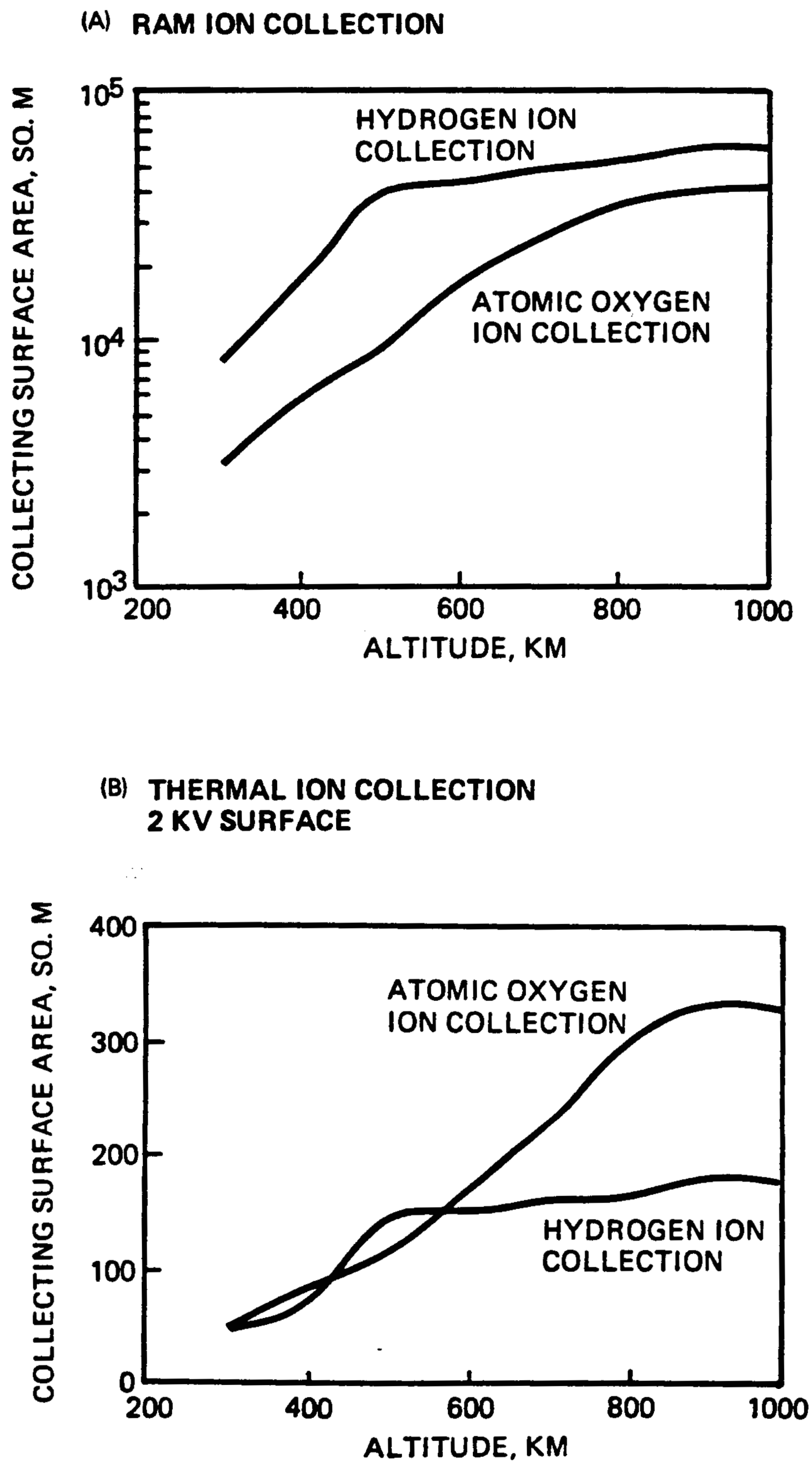

Figure 4-3(a)(b). Surface Area Required for 1 A of Current Collection

5. TETHER MATERIALS, DESIGN AND MANUFACTURE

In this section, the requirements for the tether and a detailed design addressing these requirements including the materials for construction and the engineering configuration are presented. Additionally, a report is given on the manufacturing of 1100 ft (335 m) of cable that will allow ground-based evaluation of cable performance.

5.1 REQUIREMENTS

The primary components of the tether consists of two elements, a flexible wire with low electrical resistivity and an insulating sheath that prevents electrical discharge and protects the wire from erosive or corrosive environments. Additional requirements for the wire include low mass, benign responsiveness to nonrigid body dynamics and flexibility. The insulation material must be able to avoid degradation from atomic oxygen attack in LEO and have sufficient durability to sustain mechanical handling in deployment and retrieval operations without suffering damage.

5.2 MATERIALS

Only aluminum among low resistivity conductors has the appropriate combination of high mass conductivity and mechanical properties suitable for the tether conductor. To provide the required flexibility and fatigue durability, the aluminum should be fine stranded wire that minimizes permanent deformation while undergoing dynamic gyratory deployment, retrieval and nonrigid body motions.

The insulation material must be restricted to the fluoroplastics to provide both suitable dielectric strength and atomic oxygen protection in LEO. Among the fluoroplastics, fluorinated ethylene propylene (FEP) has the desired properties and is state of the art as cable insulation.

5.3 DESIGN

In addition to the primary components required for the electrical performance of the tether, other materials of construction are necessary to provide strength and abrasion resistance in manufacture and use. The design configuration of the tether is shown in Figure 5-1. The 0.5 mm Kevlar core provides strength to the tether in use as well as strength to

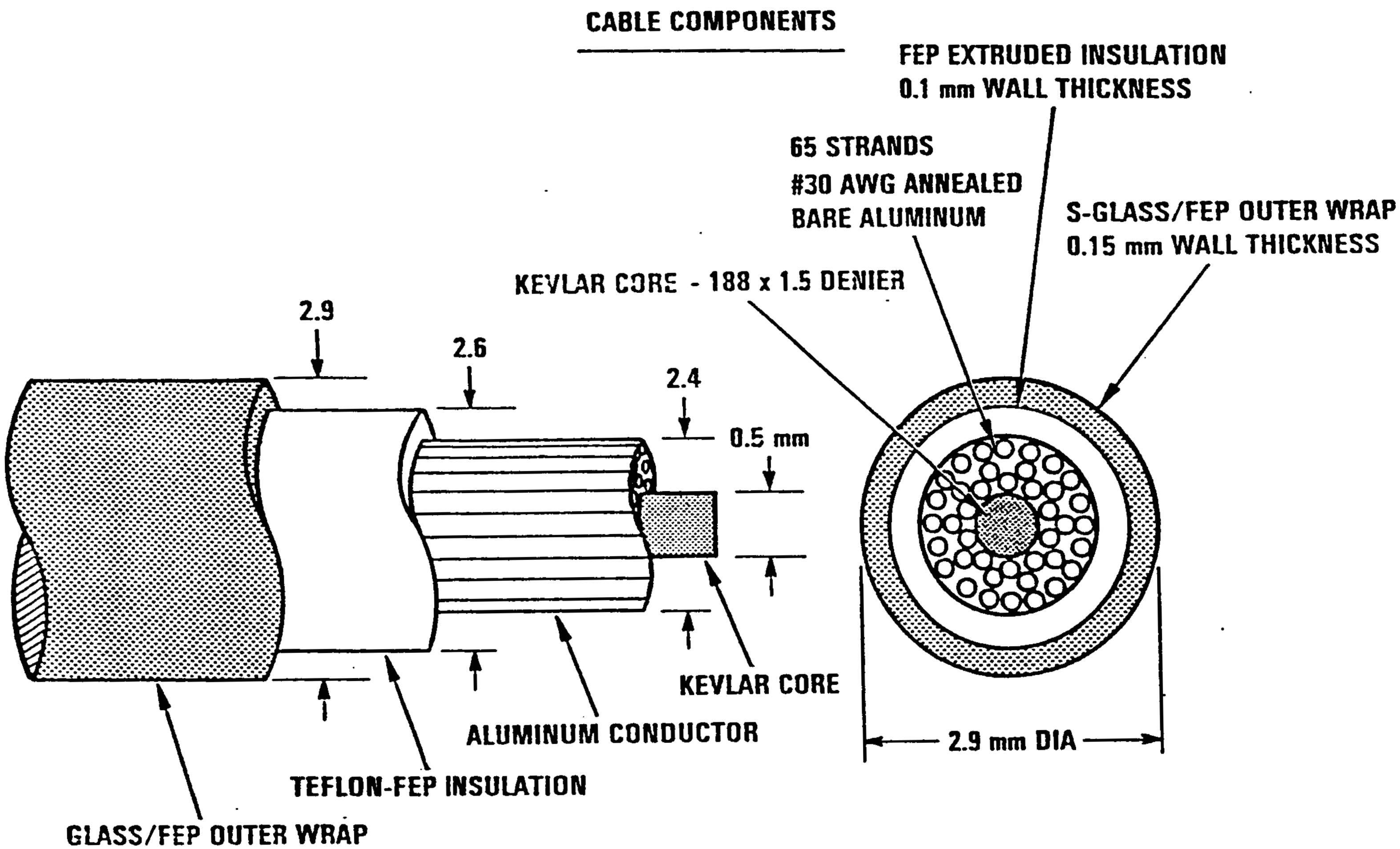

Figure 5-1. PMG Conducting Tether Configuration

the cable during manufacture. The aluminum conductor represents the major mass of the cable and dominates the mechanical response of the cable. To provide appropriate torsional balance, the strands of the aluminum are divided in approximately equivalent proportions and counter wound about the Kevlar core. A coating of FEP insulation is applied to the aluminum conductor and overwrapped with glass fiber braid which is subsequently impregnated with FEP. The glass fiber braid provides both an abrasion resistant outer coating and an additional strength component to the cable.

5.4 MANUFACTURING

In order to evaluate the tether design, a purchase order was placed with Brand-Rex Cable Systems Division of BRIntec Corporation, Willimantic, Connecticut to fabricate 1000 ft (305 m) of cable as designed. However, constraints of time and manufacturing limitations on a small batch order required alterations to the design. The cable actually manufactured is shown in Figure 5-2. The aluminum conductor consists of six strands of No. 24 AWG wire wrapped around the Kevlar core and an outer layer of fifteen strands of No. 26 AWG wire cowound about the inner layer. Two mil thick Kapton tape is counter wound about the conductor and 10 mils of FEP are extruded onto the Kapton. Major changes to the design are the addition of Kapton and the elimination of the glass braid.

The Kapton tape was introduced to protect the aluminum conductor from oxidation as it passed through the preheat furnace and hot FEP extrusion die. The glass braid contributed abrasion resistance and strength; however, these properties were redundant in the design. The Kevlar core provides sufficient strength to the cable to meet the tether requirements and the FEP insulation has sufficient toughness to withstand the single deployment and test sequence planned for the tether demonstration flight test.

If the performance evaluations of the test cable support these contentions, the alternative design will contribute two important advantges to the tether design configuration. The elimination of the glass braid represents a significant weight saving and the construction of the cable would use state of the art process technology that will be reflected in a lower cost than the original design.

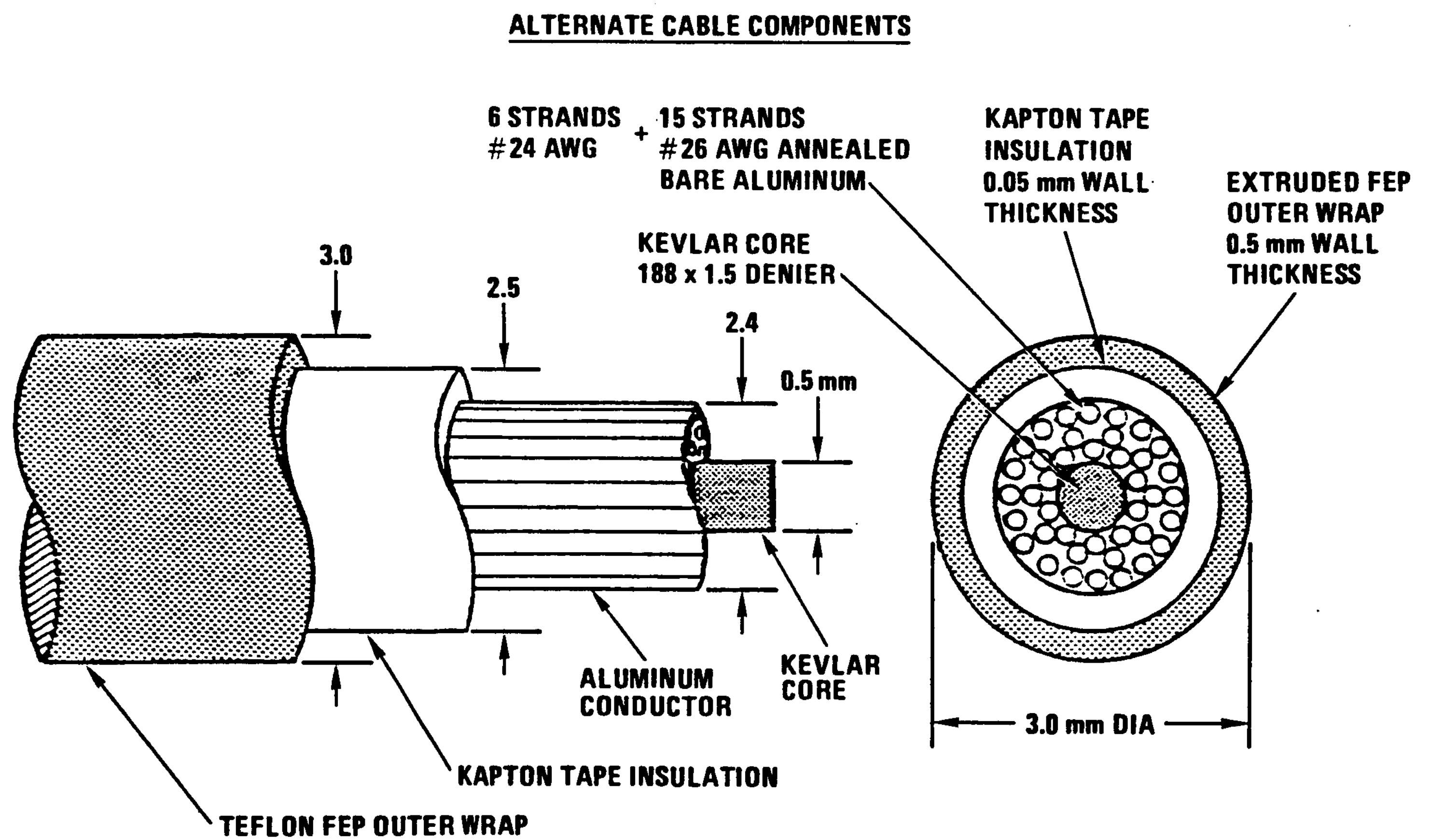

Figure 5-2. Manufactured PMG Conducting Tether Configuration

6. ELECTRONICS

Since 1969, TRW has had an advanced power system and power electronics development program in support of the future multikilowatt space missions. Extensive power electronics development has been performed on the 30 cm Mercury Ion Thruster Electric Propulsion Power Processing Equipment for NASA Lewis Research Center. Much of this thruster electronics technical background can be directly applied to the PMG Tether System Program which will greatly reduce technical, cost and schedule risks. The power electronics work will be performed by the Power Electronics Research Center in the Electrical Systems Operation.

The Power Electronics Research Center is staffed by senior power electronics design engineers specializing in space conditioning design development, packaging and system integration.

The following subsections briefly discuss the conceptual power system configuration, the power electronics designs for the different power supplies, an estimate of the power system characteristics and a brief discussion of the necessary technology programs to support the PMG Tether System Program.

6.1 POWER SYSTEM CONFIGURATION

Figure 6-1 displays the tether power system block diagram. The system includes the power electronics on board the main or host spacecraft, on the tether end mass and the interconnecting tether cable.

The host spacecraft solar array and battery is used as the prime power source for the PMG tether system. A 100 VDC power system is assumed but a 28 VDC bus could easily be used.

The power system includes a tether power supply rated at 2 to 4 kW, an anode power supply rated at 20 W and a heater power supply rated at 100 W.

The end mass assembly contains a small solar array battery auxiliary power source operating at 12 V which is used primarily to start the end mass electronics. A charger power supply is used to tap a portion of the tether power cable to charge the end mass battery and to operate the end

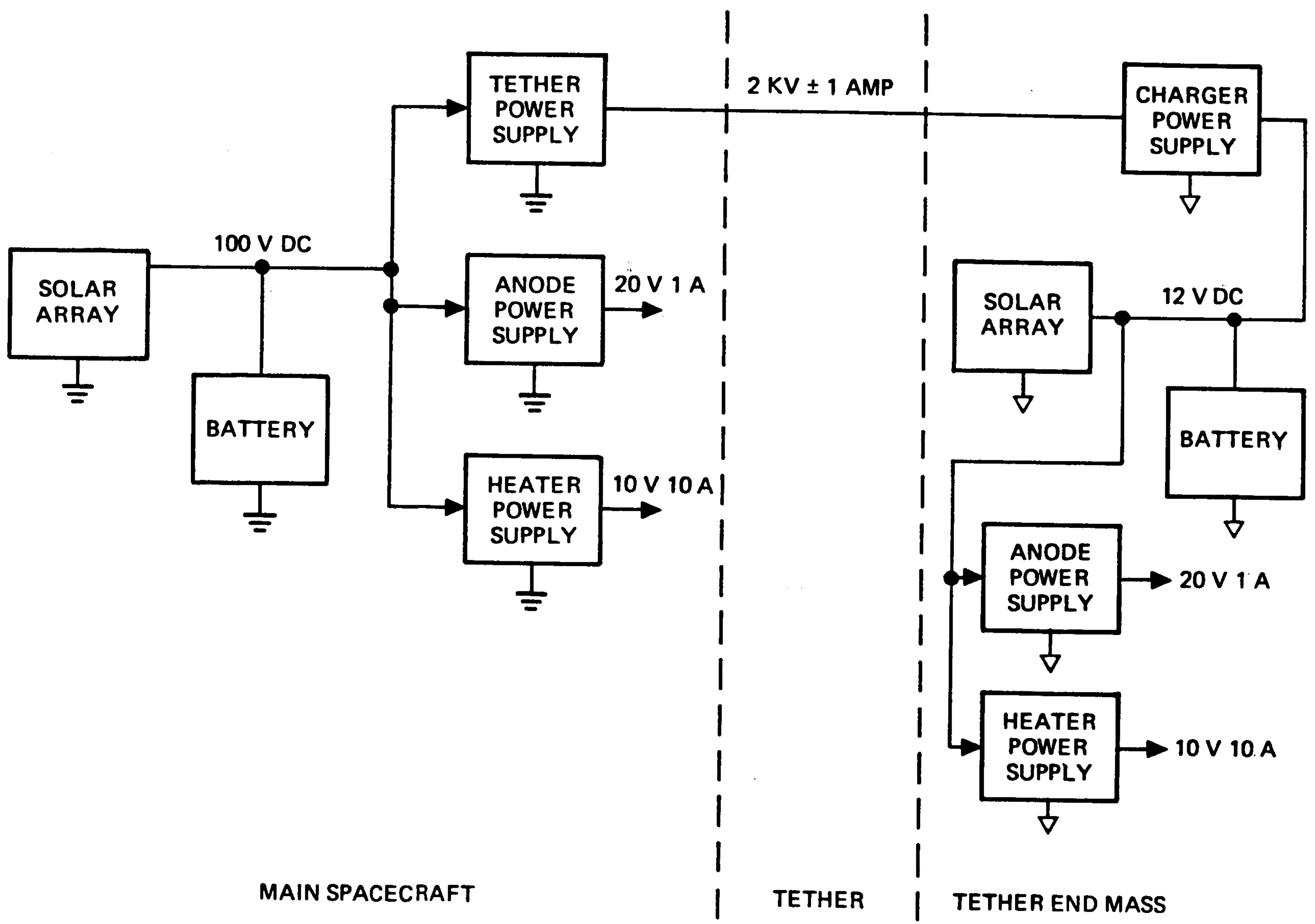

Figure 6-1. Tether Power System Block Diagram

mass power electronics. Both anode and heater supplies are used to operate the end mass plasma contactor.

All of the power supplies in the PMG tether system use the current source circuit topology that can operate in short or arc condition without any overcurrent stress condition on the power electronic components. Then electronics have been fully developed and flown on flight programs with demonstrated reliability.

6.2 TETHER POWER SUPPLY

A bidirectional DC/DC converter is proposed for the high voltage tether power supply (Figure 6-2). Bidirectionality is required to deliver current to the tether system to provide orbit boost capability or to absorb power from the tether system to provide orbit braking capability. The low voltage section (host bus voltage) uses power transistors as the switching elements in the normal power flow operation. High voltage triodes are used as switching elements (due to 2 kV requirement) during the reverse power flow operation. Inductor L1 is used during normal operation with inductor L2 shorted by the relay. During reverse power flow inductor L2 is used.

The normal power operation cycle will be briefly discussed. Assume transistors Q1 and Q2 are turned on, DC power is allowed to be stored in inductor L1 and delivered to the power transformer T1. When the control electronics has determined that sufficient energy has been delivered by the DC source, transistor Q1 is turned off, and the energy stored in inductor L1 is allowed to flow into the power transformer through transistor Q2 and diode CR4.

On the next half cycle transistors Q3, Q4 and diode CR6 perform a similar sequence to maintain output voltage/current regulation.

The output AC voltage of the power transformer is rectified by diodes CR_A, CR_B, CR_C and CR_D. During reverse power flow, inductor L2 is switched in and triodes V_A and V_B and diode CR_E act in a similar manner to transistor Q1 and Q2 and diode CR5 during normal power flow. Triodes V_C and V_D and diode CR_F operate on this alternate half cycle. Diodes CR1, CR2, CR3 and CR4 rectifies the low voltage AC winding voltage with reverse energy giving into the host power system battery.

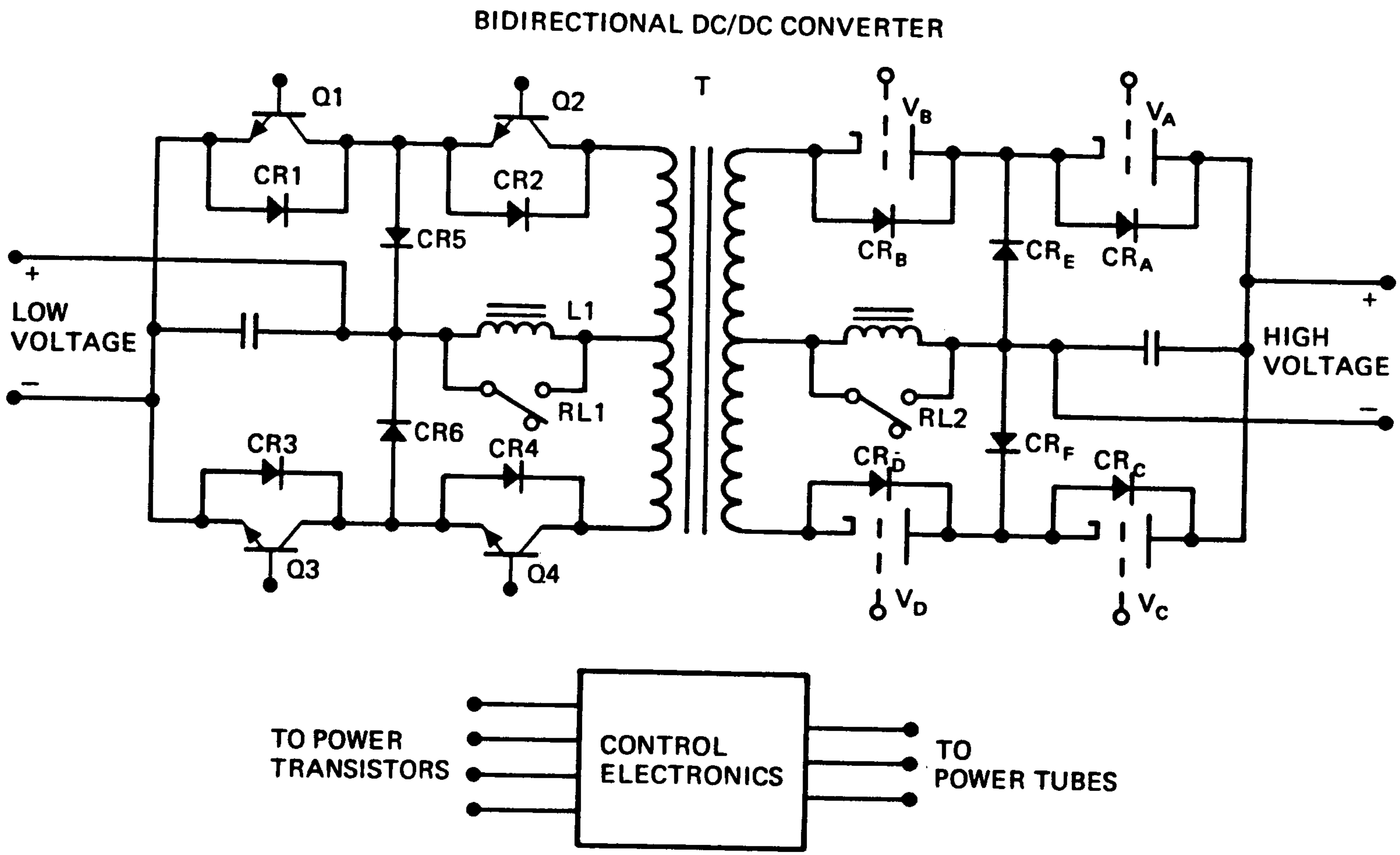

Figure 6-2. Tether Power Supply Schematic

6.3 AUXILIARY POWER SUPPLIES

Conventional current source dc/dc converters are used for the anode and heater power supplies. Figure 6-3 shows the power stage circuit topology for a 100 VDC input and Figure 6-4 displays the power stage circuit topology for 12 to 28 VDC bus voltage.

This same basic power stage is used for the end mass charger where power from the tether cable is used as the input power source and the output is used for the 12 VDC bus power and battery charging.

6.4 POWER EQUIPMENT ESTIMATES

Table 6-1 provides an engineering estimate of weight, size, efficiency and losses for the power equipment required for the PMG tether system. Both main spacecraft and tether end mass devices are estimated. The weight, size and efficiency data are derived from existing flight hardware except for the tether power supply. The solar array is estimated as body mounted cells where the total effective area is 2 ft^2.

6.5 TECHNOLOGY DEVELOPMENT ISSUES

Most of the power electronics component, circuit and packaging technology has been developed and are presently available for all of the low voltage DC/DC power supplies. The series inductor converter circuit topology development for each power supply is completed. The 2 kW power rating development was performed under a TRW IRAD. While 2 kV output voltage at 2 kW has not been demonstrated, 10 kV at 500 W has flown on the Communication Technology Satellite (CTS) and 1 kV at 2 kW has been demonstrated for ion electric propulsion. The keeper supply electronics was developed for the Ion Electric Propulsion Program as well. A limited amount of development work is required for the bidirectional tether power supply and for the tether to battery charger power supply.

There are two development issues for the tether power supply, the application of power triodes as converter switches operating at 2 to 4 kV and the associated triode heater losses and triode saturated drop. The bidirectional circuit operation has been demonstrated on the space station advanced development program for NASA Lewis Research Center.

MAIN SPACECRAFT DC/DC CONVERTER (100 V DC INPUT)

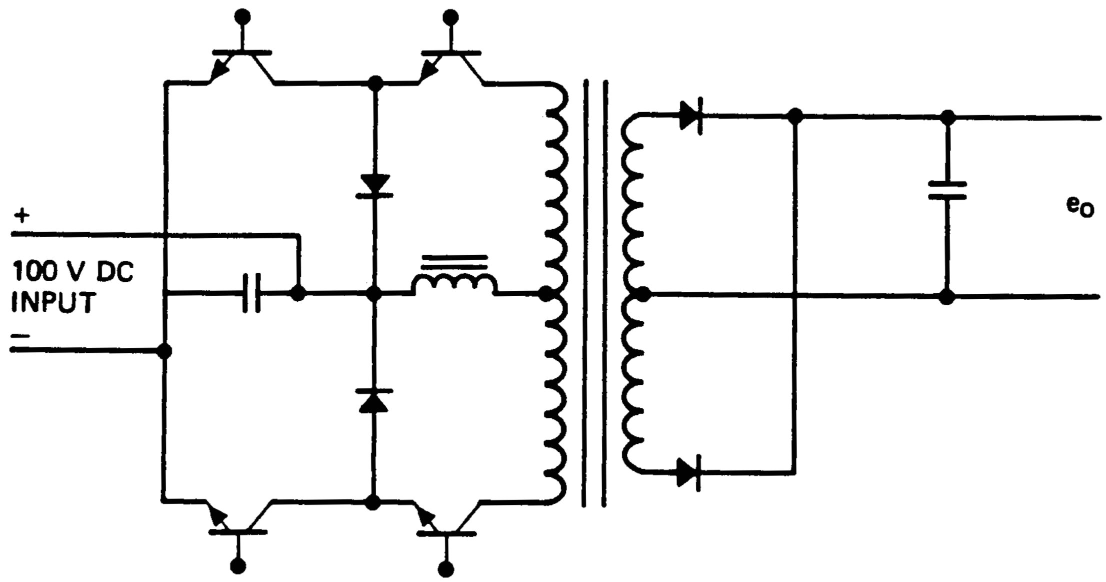

Figure 6-3. Anode/Heater Power Supply Schematic

TETHER DC/DC CONVERTER LOW VOLTAGE INPUT (12 V DC)

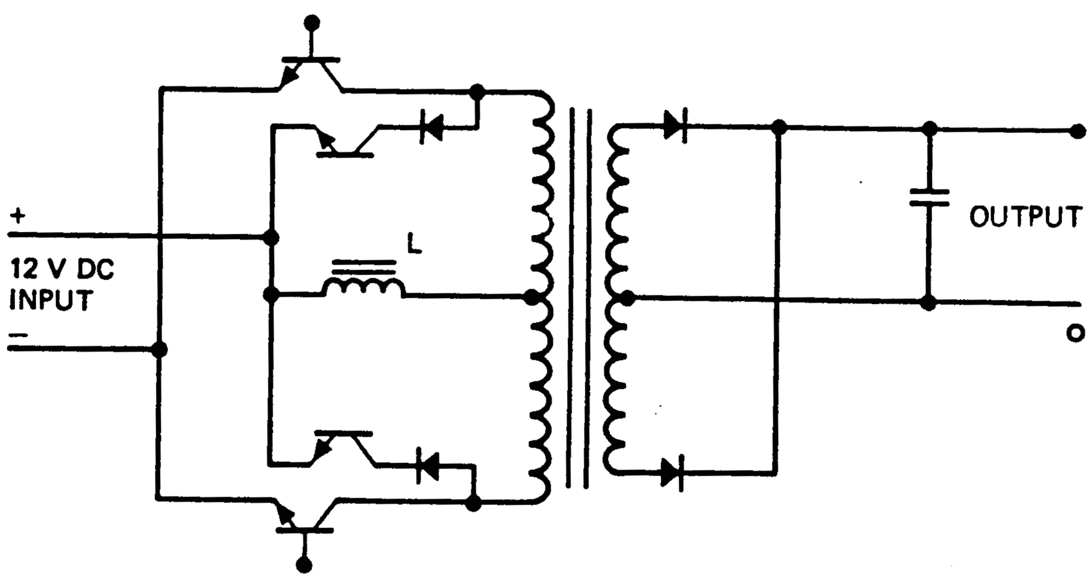

Figure 6-4. Charger-Anode-Heater Power Supply Schematic

Table 6-1. Estimates for the Power Equipment

	Weight (lb)	Mass (kg)	Size LWH in (cm)	Efficiency (%)	Losses (W)
Main Spacecraft					
Tether Power Supply (2 kW)	30	13.6	16 x 16 x 4 (40.6 x 40.6 x 10.2)	92 (Forward) 88 (Reverse)	174 273
Anode Power Supply (20 W)	1	0.4	3 x 4 x 1 (7.6 x 10.2 x 2.5)	90	2.2
Heater Power Supply (100 W)	3	1.4	5 x 5 x 4 (12.7 x 12.7 x 10.2)	88	13.6
Tether End					
Charger Power Supply (150 W)	3	1.4	6 x 6 x 4 (15.2 x 15.2 x 10.2)	88	20.5
Anode Power Supply (20 W)	1	0.4	3 x 4 x 1 (7.6 x 10.2 x 2.5)	88	2.7
Heater Power Supply (100 W)	3	1.4	5 x 5 x 4 (12.7 x 12.7 x 10.2)	86	16.2
Solar Array (20 W)	8	3.6	6 ft^2 (0.6 m^2)	-	-
Battery (12 V 10 AH)	12	5.4	7 x 7 x 3 (17.8 x 17.8 x 7.6)	-	-

The control electronics for the end mass charger is the other basic development task to ensure a reliable startup and control sequence to obtain power from the tether cable system.

7. END MASS AND DEPLOYMENT CONCEPT

In this section, descriptions of three deployment concepts for the PMG tether system are presented. These concepts are play out from the host vehicle, play out from the end mass and play out from both ends. The first two employ a static reel which is simply a tether coil container from which the tether plays out. The third uses a static reel on the end mass and a dynamic reel on the host vehicle. The impact on the end mass design will be discussed for the first two concepts. A recommendation for deploying from the end mass will be supported in this section and also in Section 11.

7.1 CONCEPT 1 HOST VEHICLE PLAY OUT

For this concept, the end mass contains only the plasma contactor assembly and thruster system. Figures 7-1 and 7-2 display the system in detail. Table 7-1 lists the weight and mass estimates for Concept 1. The advantages of locating the depolyer on the spacecraft bus include attitude stability during deployment and the facilitation of data collection and smart interactive control.

7.2 CONCEPT 2 END MASS PLAY OUT

In this case, the end mass will contain the reel and tether in addition to the plasma contactor assembly and thruster system. Figures 7-3 and 7-4 show this concept in detail. Table 7-2 presents the weight and mass estimates. Decreased initial impulse due to the large end mass is one advantage of this concept. In addition, during the deployment, the tether remains stationary relative to the spacecraft ensuring a steady, smooth play out. For these reasons, Concept 2 is the recommended configuration for deployment.

7.3 CONCEPT 3 PLAY OUT FROM BOTH ENDS

This configuration has the advantages of both of the above but is more complex than either. A detailed study of this concept was not made.

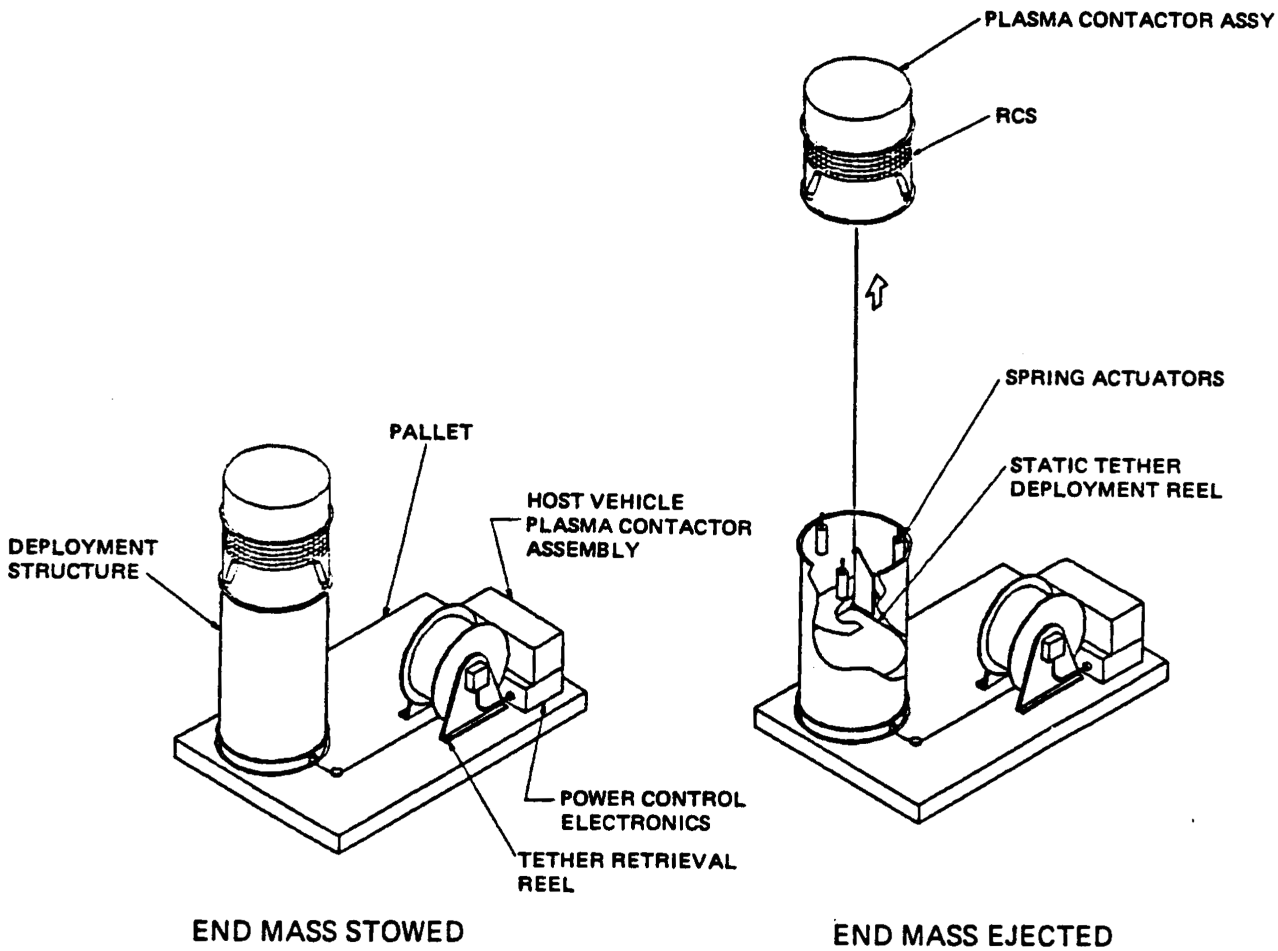

Figure 7-1. Concept 1 Tether Deployment Reel on Host Vehicle

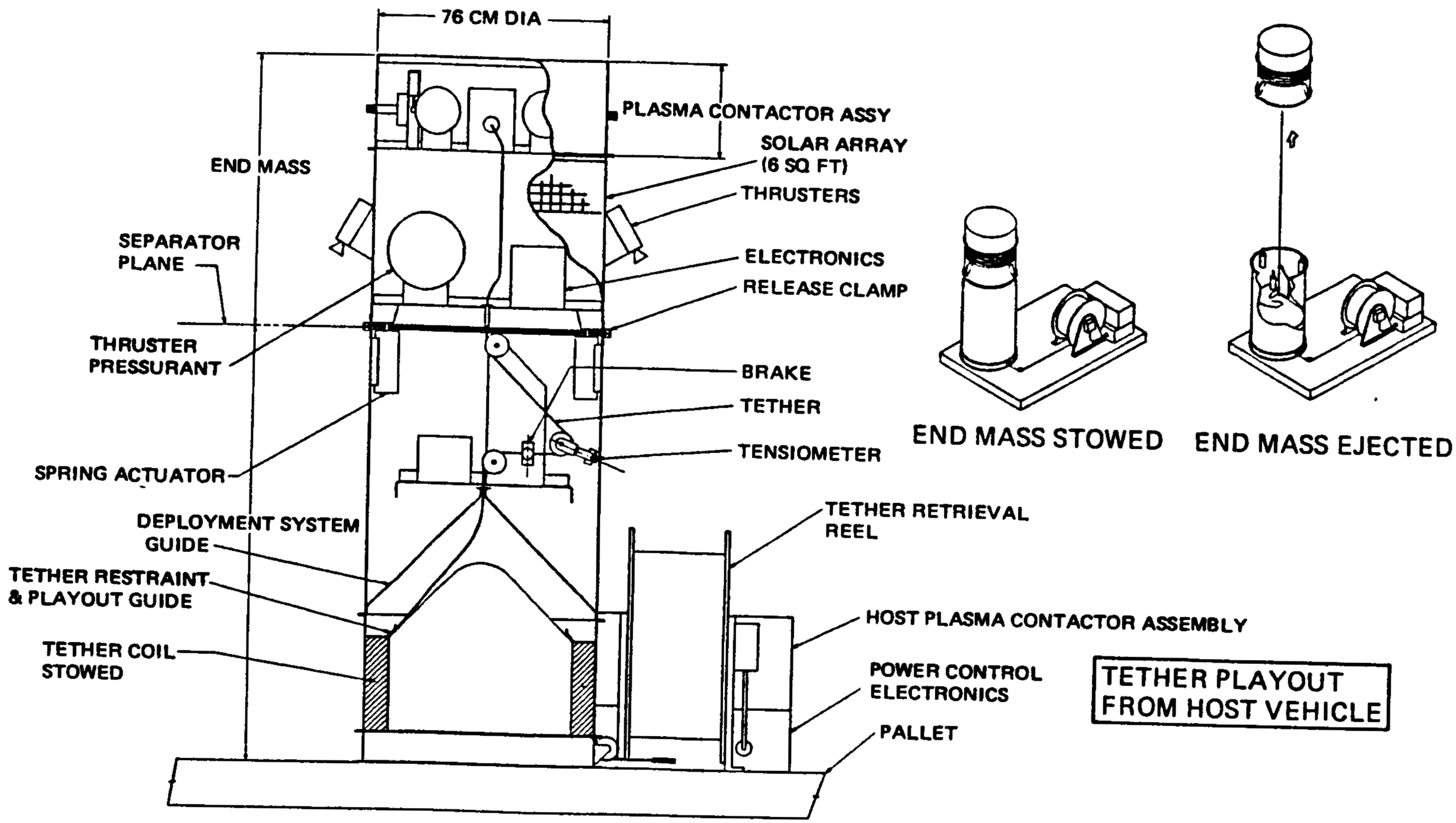

Figure 7-2. Concept 1 Tether Deployment Reel on Host Vehicle

Table 7-1. Weight and Mass Estimates for Concept 1

	Weight (lb)	Mass (kg)
End Mass		
Plasma contactor system (including power control electronics)	120	55
RCS	25	11
Cylinder structure	36	16
Total	181	82
Host Spacecraft		
Deployment structure	60	27
Retrieval reel	28	13
Host end plasma contactor assembly	93	42
Power control box	54	24
Tether (10,000 M)	306	139

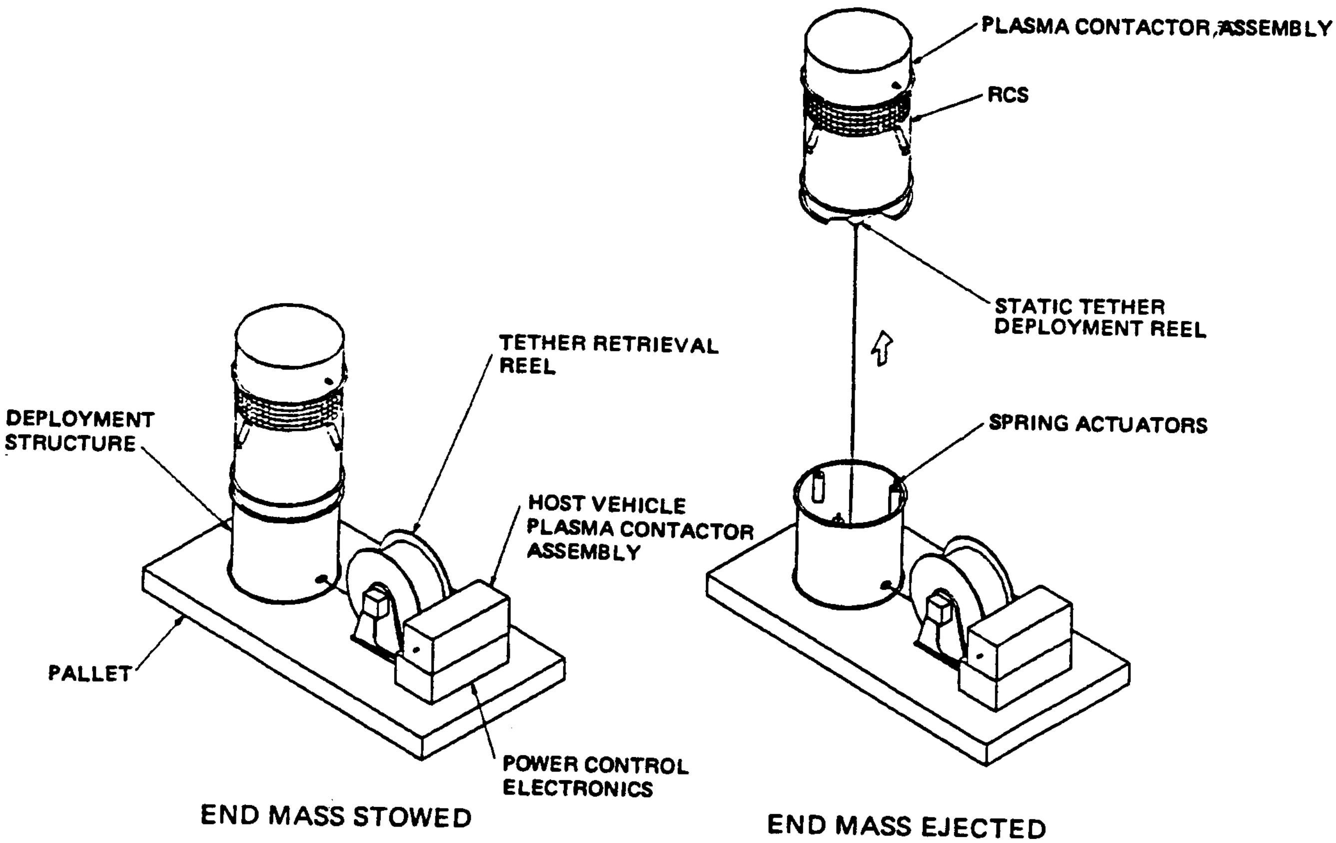

Figure 7-3. Concept 2 Tether Deployment Reel in End Mass

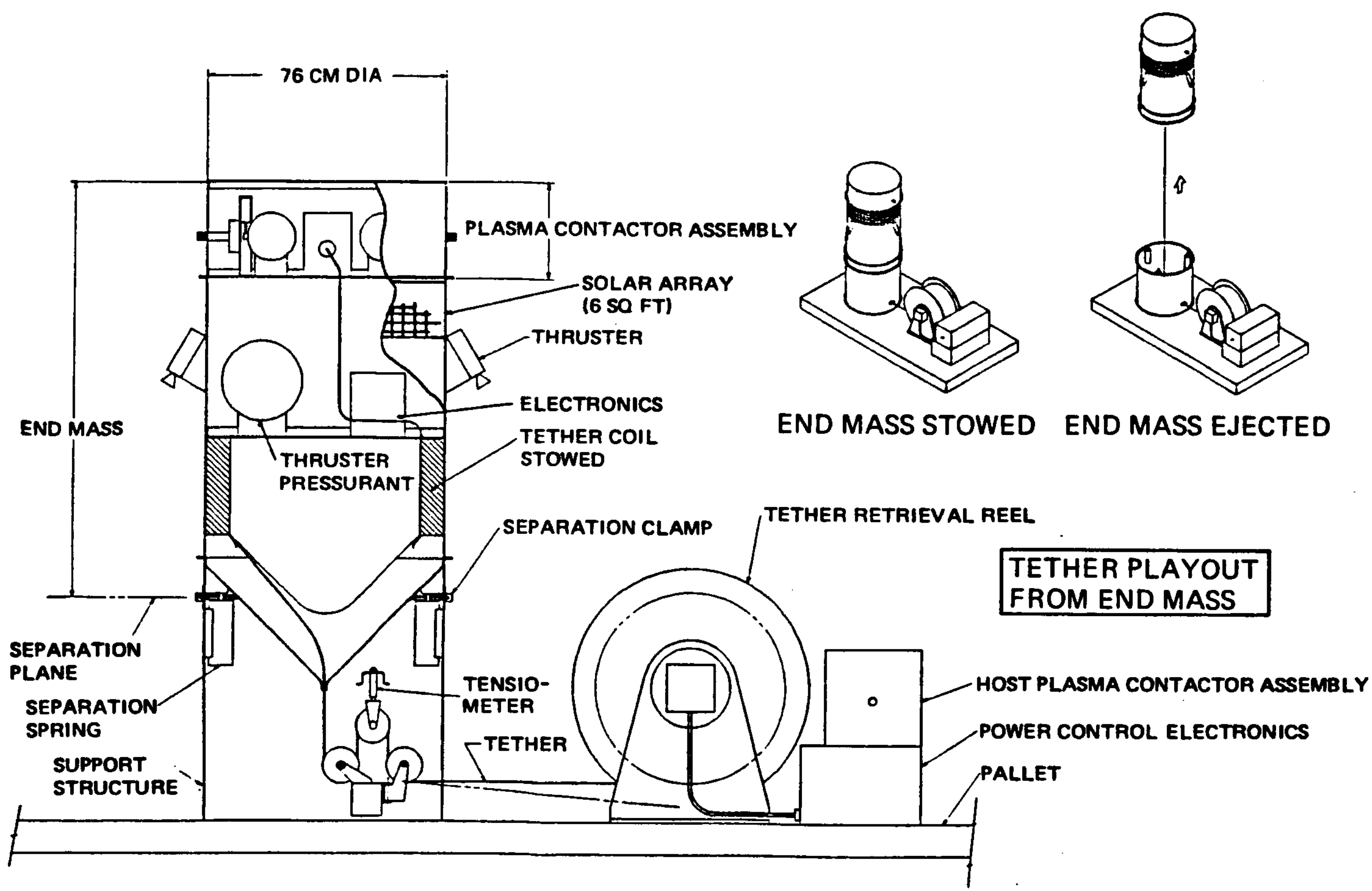

Figure 7-4. Concept 2 Tether Deployment Reel in End Mass

Table 7-2. Weight and Mass Estimates for Concept 2

	Weight (lb)	Mass (kg)
End Mass		
Plasma contactor system (including power control electronics)	120	55
RCS	25	11
Cylinder structure	52	24
Tether	306	139
Total	503	229
Host Spacecraft		
Deployment structure	24	11
Retrieval reel	28	13
Host end plasma contactor assembly	93	42
Power control box	54	24

7.4 DEPLOYMENT SCENARIO AND END MASS

The remainder of this section addresses the deployment scenario and end mass for Concept 2. To begin the deployment, an initial velocity of about 1 m/s is imparted to the end mass by spring actuators. This velocity will be sufficient to propel the end mass to a position where the gravity gradient force will sustain the play out. Cold gas thrusters will provide a backup force source.

The end mass is mated to a matching deployment cylinder containing separation springs and pulleys that route the tether to an external retrieval reel. The end mass is secured to the deployment cylinder by a releasable clamp forming a strong and simple structure well suited to resist launch loads and permitting a symmetrical application of load from the separation springs.

The static reel is an annular cavity in the end mass cylinder containing the coiled tether. A sheet metal closure at one end restrains the coil during ground handling and launch of the host vehicle. The closure is formed with a lip concentric to a central hub to create a circular slot from which the tether plays out during deployment. A one shot energy absorber is incorporated in the last few coils to reduce the end mass velocity and associated deceleration loads. The fully deployed tether will also act as a low rate spring to limit the deceleration force.

The power control electronics described in the previous section are housed within the end mass cylinder and the solar array is attached to the exterior surface.

The deployment cylinder is mounted on a pallet together with the host vehicle plasma contactor assembly, a power control electronics module and an optional tether retrieval reel. The retrieval reel would be used to return the end mass to the host vehicle after completion of the experiment thus eliminating concerns of space debris and permitting post flight evaluation of hardware should that be desired.

8. INTEGRATION WITH OMV AND EURECA

Preliminary analysis of the integration of the PMG tether system with the two candidate host spacecraft has been completed. Since EURECA is designed to be an all-purpose payload host, it appears that the only significant modification required is the use of the expanded solar array option. OMV would require the addition of a truss structure and complete power kit to accommodate the payload. Table 8-1 presents weight and mass estimates for the additional hardware for OMV. The array is sized for 3 kW of power.

Table 8-1. Weight and Mass Estimates of Additional Experiment Hardware for OMV

	Weight (lb)	Mass (kg)
Solar arrays	396	180
Truss structure	176	80
Tie downs, release mechanisms and contingency	57	26
	629	286

Both vehicles easily accept the PMG hardware components and, for OMV, the orbiter does not constrain the length of the deployer/end mass cylinder. With EURECA, the cylinder axis is normal to the orbiter x axis and is therefore constrained by the 180 inch diameter payload envelope. This does not appear to present a serious design problem, but must be addressed when selecting a deployment approach.

Figure 8-1 and 8-2 depict the PMG tether system on the OMV and on EURECA.

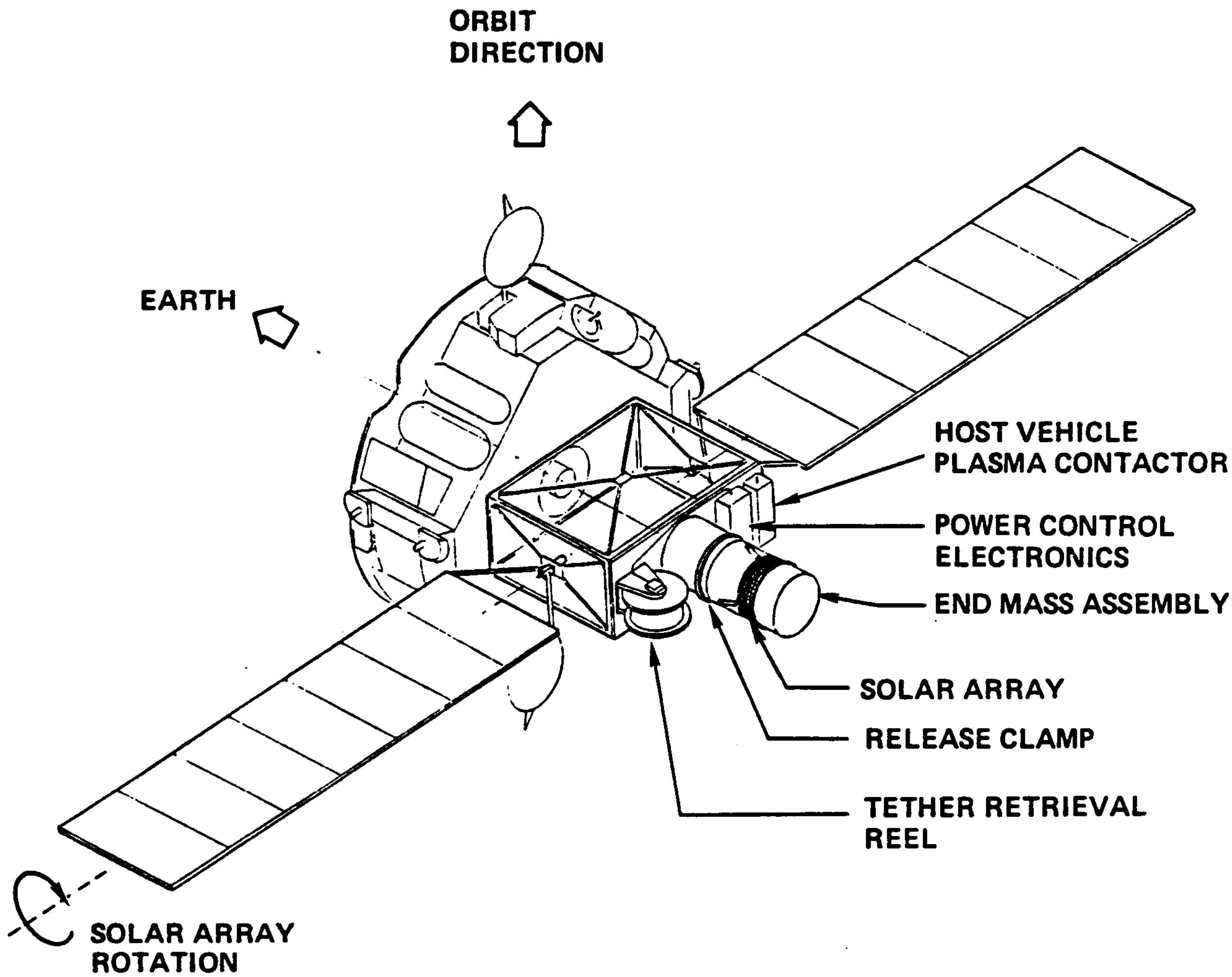

Figure 8-1. PMG Tether System on OMV

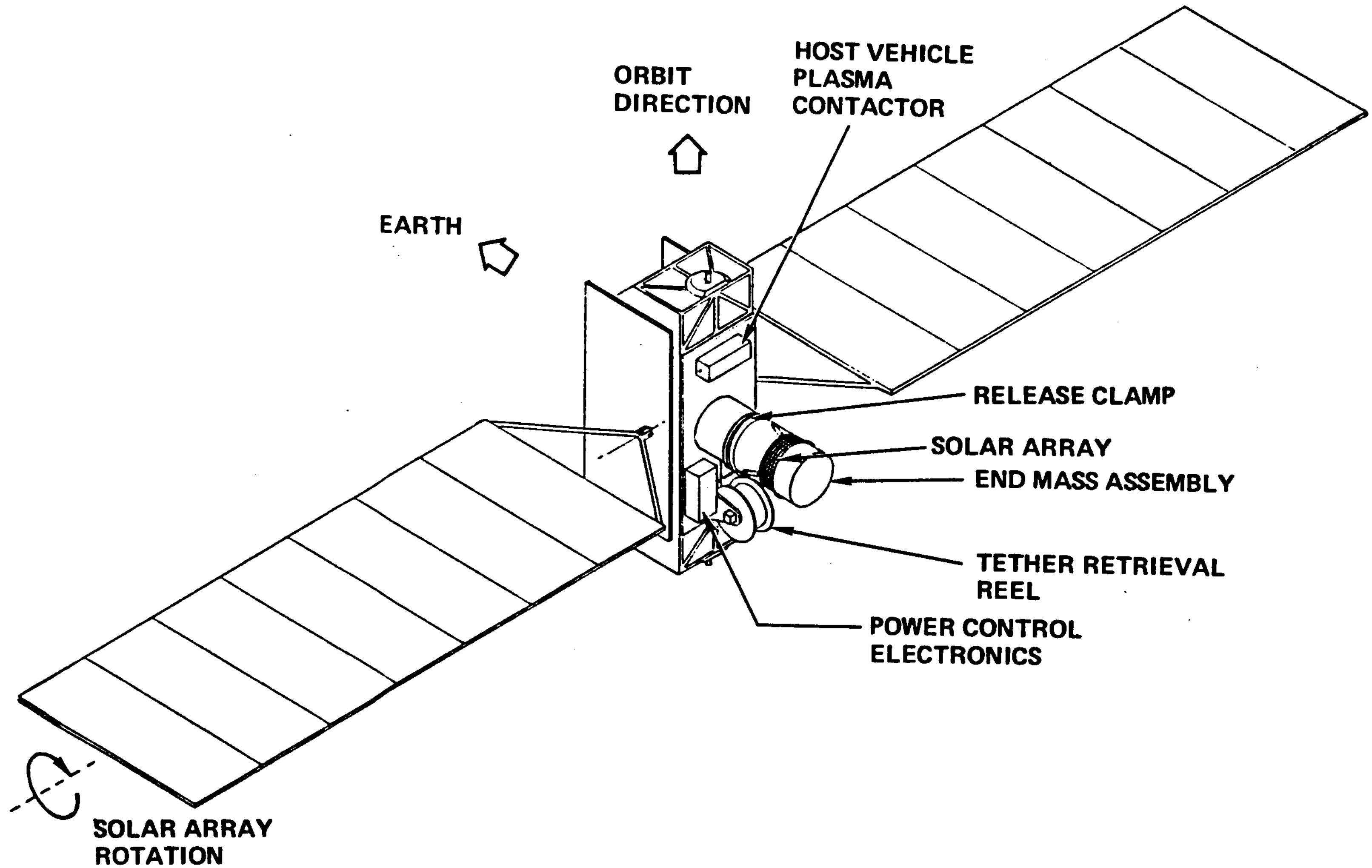

Figure 8-2. PMG Tether System on EURECA

9. COST ESTIMATES

As part of the production of performance figures, cost estimates of the hardware and project level elements defined during the study were prepared. Since the definition of the PMG tether system/host spacecraft interface remains incomplete, along with mission operations support, launch system/host spacecraft services and PMG tether system/launch system interface, it is very likely that the cost of the actual system will be higher.

Additional ground rules and assumptions for these estimates are listed below:

1. Estimates are provided for deployer in end mass configuration (Concept 2).
2. Cost stated in millions of 1987 dollars.
3. Cost given through cost of money and do not include fee or profit.
4. Cost for each item includes: project management; system engineering; design, development and hardware manufacturing; system integration assembly, test and verification; and management reserve.
5. With the exception of the OMV accommodations, 100% new design on each line item is assumed.
6. OMV accommodations cost is estimated as a procured item without design or development cost.

With the exception of the tether for which an estimate was obtained from the vendor and of launch operations support for which a level of effort was assumed, all other tether system items were estimated using the RCA PRICE H model. The OMV accommodations were estimated by analogy to TRW spacecraft. A low and a high estimate is given by line item in Table 9-1. These reflect the range of manufacturing complexity for each item.

Table 9-1. Cost Estimates for Elements of the PMG Tether System

Item	Cost ($M FY87)
Host Vehicle Hardware	
Deployment assembly	0.4 to 0.5
Plasma contactor assembly	0.5 to 0.6
Power control assembly	5.5 to 6.3
Pallet assembly	1.3 to 1.6
Tether	0.2 to 0.3
End Mass Hardware	
Deployment/structure assembly	0.8 to 1.0
Plasma contactor assembly	0.5 to 0.6
Power control assembly	1.2 to 1.4
Electrical power assembly	1.0 to 1.1
Reaction control system assembly	6.1 to 7.0
Launch Operations Support	0.0 to 1.0
Total	17.5 to 20.5
OMV Accommodations	2.5 to 3.5
Host Vehicle Retrieval Assembly	0.3 to 0.4

10. ENVIRONMENTAL CONCERNS

A study was performed to determine the impact of the meteoroid and debris environment on the performance of the PMG tether system. Two concerns were considered: the number of impacts sufficient to sever the tether per year and power losses due to penetration of the tether insulation resulting in exposure of the high voltage to the ambient plasma.

For the tether in Figure 5-1 the total surface area is 91.1 m^2. An orbital altitude of 400 km was assumed. This altitude is one of the few for which a debris environment specification has been defined. The environmental definitions are from the NASA model for meteoroids (Reference 11) and a modified version of the Space Station debris model (Reference 12). The modification imposed on the debris model was to increase the number of particles by a factor of 11 (Reference 13). The debris environment is altitude dependent while the meteoroid environment is considered to be constant.

It was determined that a particle with mass greater than or equal to 10^{-4} gm could penetrate the tether all the way through. Based on the area of the tether and the distribution of particles, it was determined that the number of impacts capable of severing the tether is 0.3 per year for meteoroids and 0.5 per year for debris.

During the period of 1 year, the number of penetrations in the insulation would be about 10 from meteoroids and up to 275 from debris particles. Assuming a uniform distribution of the 285 holes, the power loss for the 2 kW system would be about 12 W or less than 1%. There could be arcing in the regions of the tether that operate above about 800 V.

11. DEPLOYMENT SIMULATIONS

In this section, the results of deployment simulations performed at the Energy Science Laboratories, Inc. using their computer simulator, BEADSIM, are presented. The issues investigated were the location of the deployer, deployment scenarios, starting methods and deceleration at the end of the deployment. The recommendations proposed in this section were incorporated in the design of the end mass and deployer for Concept 2 in Section 7.

11.1 DEPLOYMENT ISSUES

In addition to the reasons already given, locating the deployer in the end mass results in several other desirable features. For one, due to inertial effects, the tether tension during deployment is less on the deployer end than on the other end. If the deployer is in the end mass, the force required to decelerate can be far less. Also, the kinetic energy to be absorbed at the end of deployment is only due to the end mass emptied of the tether, since the tether is not moving away from the host vehicle. Finally, the tension during deployment can be measured more easily at the fixed end than at the deployment end. Since the fixed end is the spacecraft, monitoring is facilitated.

The only disadvantage to putting the deployer in the end mass is that low tip-off rates become more important to prevent the tether from wrapping around the end mass as it unreels. Tip-off rate refers to the rotation of the end mass in inertial space. No problem is expected for tip-off rates less than a few degrees per second.

A high impulse deployment method started by springs is recommended to ensure a smooth unreeling. Spacecraft thrusters under ground control can be used if a jam occurs. Further testing is needed to determine the tension during deployment. To decelerate at the end of the deployment a one-shot energy absorber, perhaps bonding the last turn on the core, is recommended.

11.2 BEADSIM SIMULATIONS

BEADSIM was developed to simulate the Small Expendable Deployment System (SEDS) currently under study for NASA. It does not model tension variations along the tether which can be large. It can incorporate detailed data on the wire properties. The parameters studied include the deployment tension, deployment direction, packing fraction which is the ratio of the area of the core to the outer diameter of the stored tether and the final braking tension.

The results of these studies are that a 45 degree angle of deployment is best but any angle between 0 and 90 degrees is acceptable. An initial velocity of 1 m/s is recommended for the deployer located on the end mass. The range rates at the end of deployment will be 2 to 8 m/s depending on the deployer. Typical deployment time for a 10 km tether is less than 1 orbit.

Graphic output from four representative BEADSIM simulation runs for the PMG tether deployment constitute Figures 11-1 through 11-4. These "multiple exposure pictures" or "walking plots" show the tether shape and position in a rotating local vertical local horizontal reference frame at 2-minute intervals during deployment. Each curve is displaced 0.4 km to the left, the direction of orbital motion. The four cases displayed are:

1. Baseline Case: deployer on end mass, 45 degree angle of deployment, small core, 65 J spring
2. Baseline case but with large core
3. Baseline case but with upward deployment requiring a 102 J spring
4. Baseline case but with OMV-mounted deployer requiring a 260 J spring and more braking.

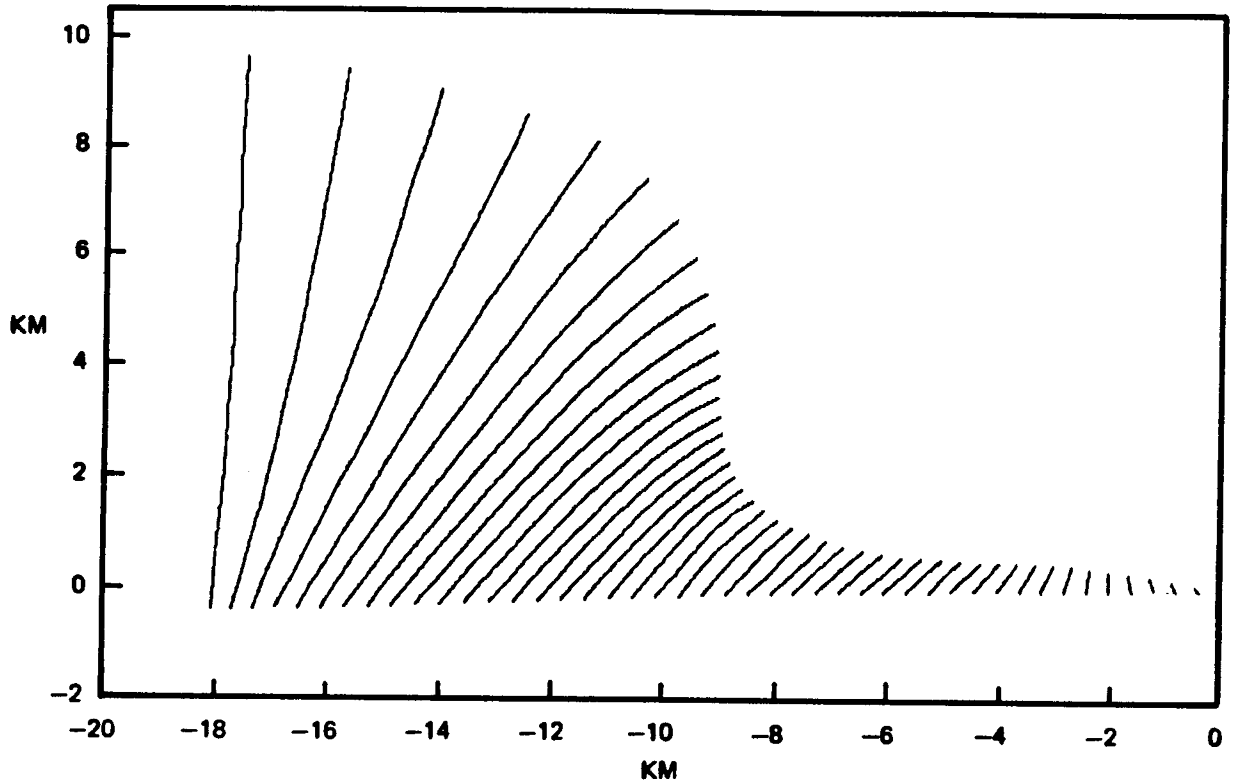

Figure 11-1. Walking Plot for Baseline Case: Deployer on End Mass, Deployment Up and Forward (45 Degrees), 65 J Spring and Small Core

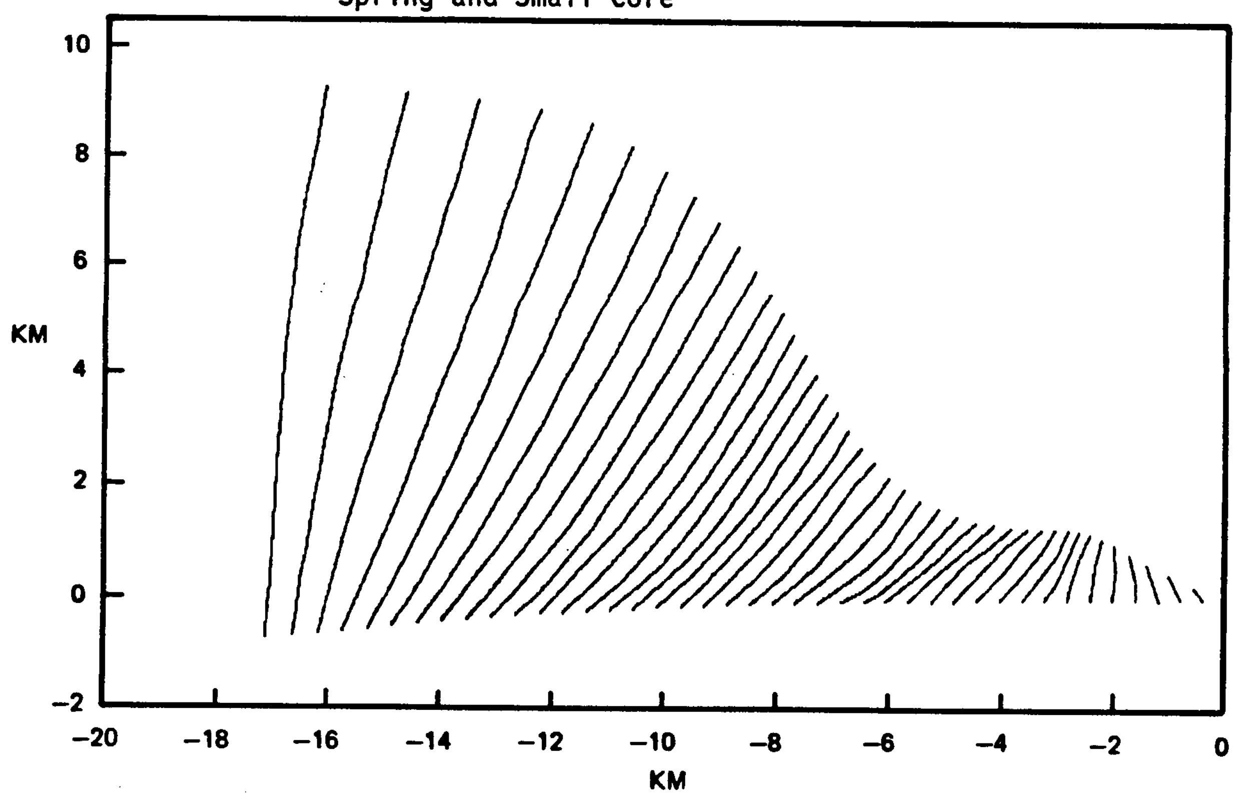

Figure 11-2. Walking Plot for Baseline Case with 260 J Spring and Deployer on OMV

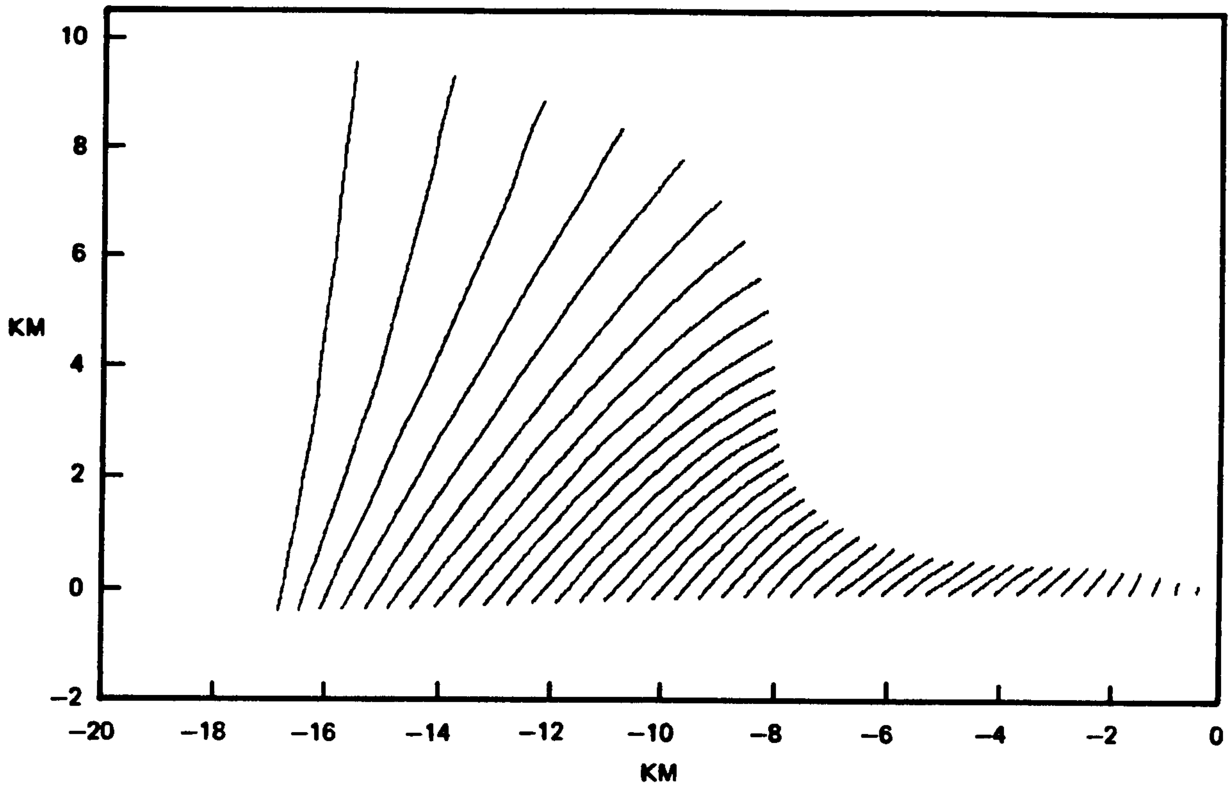

Figure 11-3. Walking Plot for Baseline Case with Upward Deployment Requiring a 102 J Spring

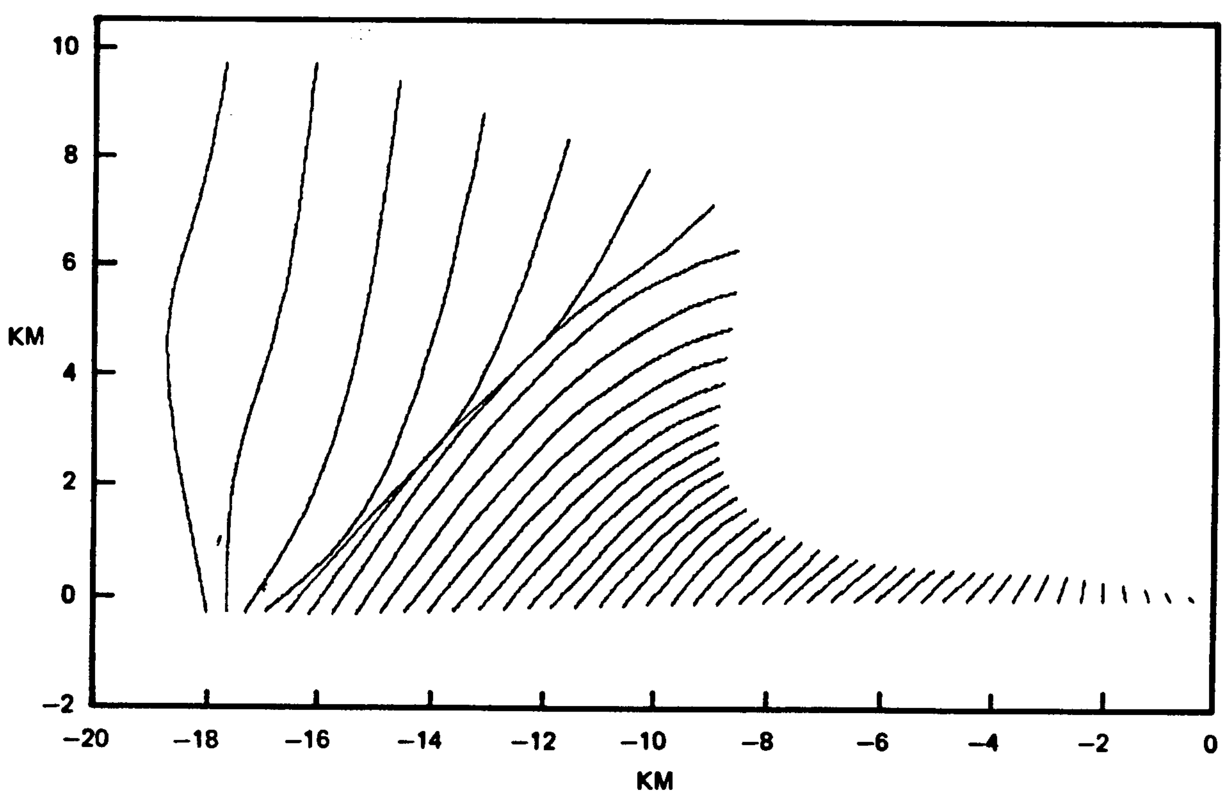

Figure 11-4. Walking Plot for Baseline Case with Large Core Requiring Much More Braking at the End

12. ENGINEERING DEVELOPMENT PROGRAM

The engineering development program consists of a two-part plan, one for technology development and one for systems requirements and conceptual design. The PMG tether system design influences the scope of the technology development. Conversely, the progress of tether technology influences the actual requirements and conceptual design. The goal is to maximize the use of off-the-shelf subsystems and hardware to ensure low cost and reduced technical risk.

A schedule and tasks for the engineering development program are shown in Figure 12-1. The first phase provides time for a thorough engineering understanding of the tether system design and its interactions with OMV and EURECA. The result will be a deeper knowledge of all spacecraft and PMG interfaces. This will also permit a full validation of system requirements and of the operations concepts.

The preliminary design and critical design phases are conventional disciplines. These phases will yield a design that meets requirements and is producible. Subsystems development and test of the six elements of the PMG tether system proceed in parallel. Assembly of the PMG tether system from the subsystem elements is accomplished and the system is integrated to the host spacecraft/platform, (e.g., OMV). Prelaunch, launch, and on-orbit operations are short-term, intensely active and critical activities. They culminate the entire effort by demonstrating the PMG tether system in a motor mode for orbit reboost.

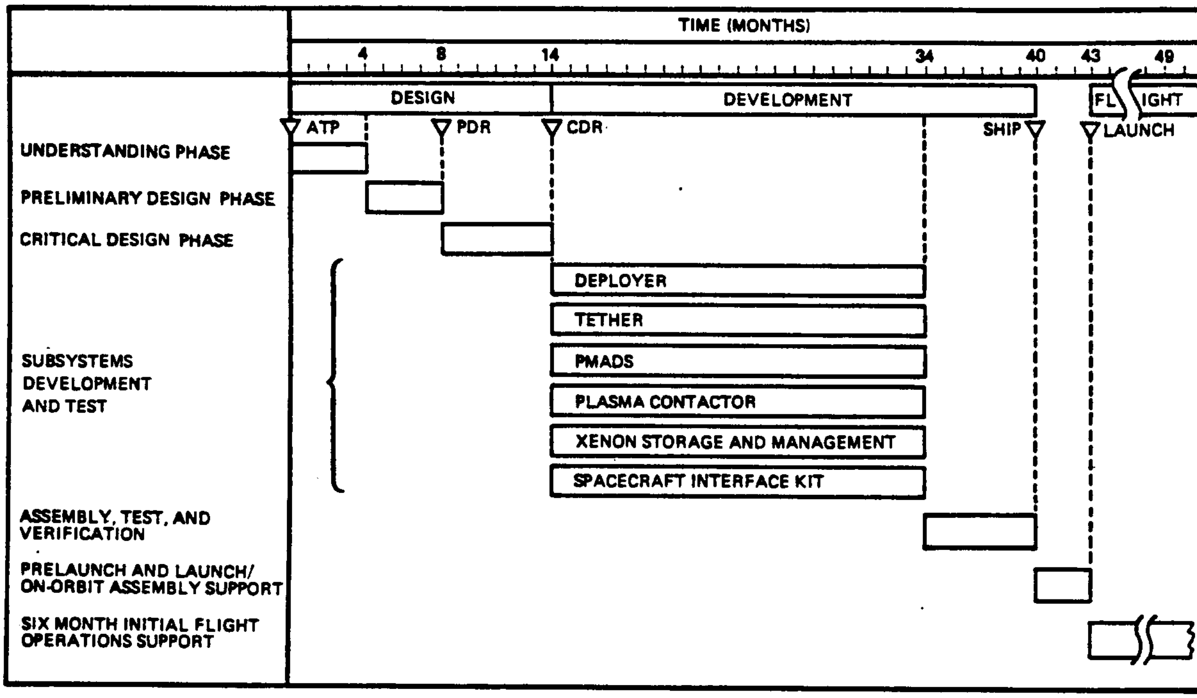

Figure 12-1. Engineering Development Schedule

13. CONCLUSIONS AND FUTURE WORK

This study has produced detailed designs for a 2 kW PMG Tether System based largely on existing hardware and hardware designs. Specifically, the hollow cathode design and electronics are derived from ion propulsion equipment. A prototype tether has been constructed and, during planned future study, will undergo deployment tests and tests for strength, resistance to breakage and abrasion and electrical properties. In addition, laboratory development models of the electronics will be used to operate two PMG hollow cathode assemblies with this tether to verify electrical performance parameters for the complete system.

Based on the dynamical properties gathered during the test of the prototype tether, refined simulations of the deployment using BEADSIM and of long term stability characteristics using the current version of GTOSS will be performed.

The mission selection process yielded five spacecraft or missions to host the flight demonstration of the PMG tether system. Of these, OMV and EURECA are the most promising. More detailed electrical and mechanical integration studies will be performed as part of the follow-on activities. Based on the results of these studies, an update cost estimate will be generated.

The results presented in this report show that a low cost demonstration of a PMG Tether System appears to be feasible by the middle of the 1990s. The potential for this innovative technology as a supplement to both propulsion and electrical subsystems for some future spacecraft and missions clearly justify such a demonstration.

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